NASA-CS(06)

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Report No. M-6

A STUDY OF INTERPLANETARY SPACE MISSIONS

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A STUDY OF INTERPLANETARY SPACE MISSIONS

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for

Lunar and Planetary Programs
National Aeronautics and Space Administration
Washington, D. C.

Contract No. NASr-65(06)

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March 15, 1965

ACKNOWLEDGEMENTS

The author gratefully acknowledges the contributions, comments and criticisms of the staff of Astro Sciences Center, IIT Research Institute.

SUMMARY

A STUDY OF INTERPLANETARY SPACE MISSIONS

The Astro Sciences Center of IIT Research Institute is performing a series of solar system exploration studies for NASA under Contract No. NASr-65(06). This report completes a survey of interplanetary space exploration which was initiated with a previously published investigation of the scientific objectives. The gross requirements for interplanetary missions are presented here and four mission profiles are provided by way of examples.

Interplanetary space can be considered as extending from the solar corona to an undefined boundary of the solar system and is all inclusive except for the relatively small volumes occupied by the planets and their near environments. It would be impracticable at this time to consider missions to embrace the whole of the solar system in view of the limited knowledge which exists even in the region of the Earth's orbit. The region of space which has been considered in most detail here is between heliocentric radii of 0.5 AU and 5 AU and heliocentric latitudes of $\pm$ 50°.

The scientific objectives of the missions are based on previous reports and include measurements of the solar wind,

interplanetary magnetic field, cosmic rays and solar protons, the distribution of interplanetary matter and to some extent electromagnetic radiation. The missions should be considered as exploratory since they may be the first to penetrate these regions of deep space. From the scientific objectives a basic experimental payload has been derived which contains two plasma probes, two magnetometers, cosmic ray detectors for particles in the range 10 to 1000 meV and 0.1 to 10 meV, an ionization chamber, a micrometeorite detector and a radiometer. One each of the plasma probes and magnetometers is required to have a time response of the order of milliseconds and represents an advance in the state of the art. The same basic experimental package has been assumed for each mission with similar data storage and sequencing equipment. Minimal guidance propulsion has been included sufficient only to correct launch errors. The main differences between the final payloads are in the communications requirements and in the types and outputs of the power supplies.

The flight parameters for the missions have been selected from families of minimum energy deep space trajectories and enable a good approximation to be obtained for the time of flight, minimum ideal velocity, communications distance along the trajectory, time above the desired latitude, heliocentric flight angle and the inclination of the orbit plane for any target position (r, β), where $0.5\ \mathrm{AU} < r < 5\ \mathrm{AU}$, and $0° < \beta < 50°$.

The most pertinent of the mission constraints is the selection of a trajectory of high scientific utility which must be complemented by compatibility with launch vehicle performances and total spacecraft weight. For most missions there will be no launch window constraint and no need for midcourse guidance. It is also found that the communications system, the power supply and the anticipated reliability are independent of the inclination of the orbit plane but critically dependent on the heliocentric distances involved.

The launch vehicles assumed for the missions are the Atlas Centaur and the Saturn 1B. However in most cases additional kick stages are required to impart the high ideal velocities to the relatively small spacecraft. The Saturn V vehicle has not been considered in detail due to the small spacecraft weights involved which would require three or more additional stages to make full use of its performance. In fact it is very probable that a thrusted stage launched by a Saturn 1B will offer advantages for interplanetary missions to large heliocentric latitudes.

Mission profiles have been developed for four missions and are summarized in the table at the back of the summary. However these profiles are by no means exclusive and may even be considered more as examples. They all use minimum energy trajectories and include an ecliptic mission to 5.2 AU representing the next phase of ecliptic exploration beyond Mariner and Pioneer. Two missions have target areas specified in latitude only (13.5° and 22°) and should be valuable for exploration missions out of the ecliptic plane. A profile is presented for a mission to a latitude of 15° and above the center of the asteroid belt. Finally suggestions are made for constructing multiple missions where the value lies in obtaining data simultaneously from two or more predetermined positions in space. These may add considerably to the present knowledge of the propagation of irregular zones through interplanetary space.

SUMMARY OF MISSION PROFILES

Mission	Target Area Dist. r AU	Target Area Lat. β°	Time of Flight TF days	Ideal Velocity ΔV ft/sec	Inc. of Orbit i°	Comm. Dist. RC AU	Payload Weight lbs.	Possible Launch Vehicle
Ecliptic to 5.2 AU	5.2	0°	500	55,000	0	4.2	360	Atlas Agena + H.E.
Absolute Min. ΔV to $\beta = 13.5°$ Lat.	0.9	13.5	90	47,000	13.5	0.25	240	Floxed Atlas Centaur
Absolute Min. ΔV to $\beta = 22°$ Lat.	0.85	22	90	56,000	22.01	0.5	260	Atlas Agena + H.E.
Min. ΔV to r = 3 AU, $\beta = 15°$	3	15	265	61,500	17.2	4	350	S1B Centaur + L.E.

H.E. = High Energy Stage (I_{SP} = 455 secs)

L.E. = Low Energy Stage (I_{SP} = 300 secs)

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A STUDY OF INTERPLANETARY SPACE MISSIONS

1. INTRODUCTION

Interplanetary space can be considered as extending from the solar corona to an undefined boundary of the solar system and is all inclusive except for the relatively small volumes occupied by the planets and their near environments. This volume of space must include the orbit of Pluto with a semi-major axis of almost 40 AU. It will contain the same basic types of particles and fields throughout, although the particle configurations and the field patterns may be grossly different in different regions of this volume. It would be impracticable at this time to consider missions to embrace the whole of interplanetary space in view of the limited knowledge which exists even in the region of the Earth's orbit, a knowledge which alone must be used as the basis for the design of the missions. In this report the region of space which has been considered in most detail is between heliocentric radii of 0.5 AU and 5 AU and between heliocentric latitudes of 0° and 50°. There is a great deal of exploratory and confirmatory investigation to be accomplished within this region, and it is on the more complete knowledge of this region that the more extensive missions should be based.

The scientific objectives of missions within the above artificially imposed but realistic limitations are not directed at just mapping the properties of interplanetary space. By coupling specific measurement techniques with a judicious choice of spacecraft trajectories it is hoped that the overall structure of particles and fields in space can be deduced. Naturally non-ecliptic trajectories have been included in this mission study. The principal measurements in all regions of space considered will be basically the same, with a high degree of correlation required between the individual particle and field measurements. Although the sensitivity, range and exposed experimental area may differ from mission to mission it is still possible to consider one basic experimental payload in a basic spacecraft structure with the major variants between the missions being the weights of the transmitter and antenna and of the power supply. With this in mind an effort has been made to keep the weight of the basic spacecraft to a minimum and hence preserve the maximum versatility in the choice of trajectories within the capabilities of current or near future launch vehicles.

The important mission parameters for each mission are determined largely by consideration of the trajectory and to a lesser extent by the temperature environment and collision hazard. For each mission considered the minimum energy trajectory is presented as it appears in the orbit plane. This allows the relative positions of the spacecraft, Earth and Sun to be rapidly deduced throughout the flights. Similarly multiple mission trajectories are shown in this way and particularly for ecliptic missions it is possible to see the launch separation and the trajectory parameters required to align the spacecraft at specified heliocentric distances.

It is logical when considering deep space missions to want to combine interplanetary and planetary investigations. This is certainly possible for some ecliptic missions and is not impossible for some non-ecliptic missions which will of course cross the ecliptic plane. However the basic experimental payload which has been derived is considered necessary for a balanced investigation of the interplanetary medium and it should not be compromised by partial replacement with planetary instrumentation. A combination of the total interplanetary payload with a planetary one however could be rewarding in spite of the restrictions necessarily placed on the missions by way of launch windows and guidance and control. In this study an ecliptic mission to the region of Jupiter has been used as an example but the only additional equipment is a radiometer which can measure Jovian radiation from large distances, i.e., $\simeq 0.25$ AU from the planet. This does not impose a launch window or guidance problem. Otherwise this study has dealt with the purely interplanetary aspects of deep space missions.

2. SCIENTIFIC OBJECTIVES OF INTERPLANETARY MISSIONS

The scientific objectives of interplanetary space missions have been studied in detail in previous ASC/IITRI reports (Roberts 1964a,b). It has been shown that the knowledge of particles and fields in the ecliptic plane, although in excess of that in non-ecliptic regions, is still very far from complete. The reason for the predominance of ecliptic data is that all scientific spacecraft whether Earth orbiting or interplanetary have been confined to the ecliptic plane. The large quantity of data which have been obtained have not provided a conclusive understanding and it is felt that thought should now be given to missions designed specifically to unravel the intricate pattern of particles and fields over a large volume of the solar system, ecliptic and non-ecliptic. In addition, more sophisticated measurements than have been made, will be required in interplanetary space with particular reference to correlation between the particle and field measurements and to detection of the fine structure of the interplanetary magnetic field and the solar wind flux.

The scientific objectives of and the measurements required in interplanetary space are summarized in Table 1.

The speed and direction of the solar wind flux has been monitored in the ecliptic plane and approximate energy spectra have been taken (Sonnett 1963, GSFC 1964a). Inadequate data is available to predict the changes in the solar wind parameters as a function of heliocentric distance or latitude, to determine the ultimate fate of the solar wind proton and electron plasma, or to fully understand the nature of the quite large and continual fluctuations in the solar wind (Lust 1963).

Table 1

SUGGESTED MEASUREMENTS IN INTERPLANETARY SPACE

Component	Major Objectives	Measurements
Solar wind	Mode of propagation from Sun. Interaction with magnetic field. Interpretation of fluctuations. Estimation of ultimate fate.	Speed, direction, flux, temperature, energy spectrum of solar wind as a function of time, position and magnetic field.
Magnetic field	Basic structure of magnetic field in space. Interaction with solar wind. Interpretation of variations in direction.	Intensity and direction relative to time, position and solar wind motion.
Cosmic rays		
Galactic	Verify extent of isotropy. Determine spatial extent of modulations. Energy and mass spectra. Flux gradient.	Energy, flux, mass/nucleon and direction as a function of time and position with respect to the Earth.
Solar	Mode of propagation. Spread of solar proton "beam". Determine full energy spectrum. Establish spatial extent of delayed isotropy of flux. Regions of minimum flux.	Energy, flux and direction as function of time and position with respect to the flare site and Earth.
Interplanetary matter	Measure direction and spatial distribution. Detect replenishing sources. Classify types of particles.	Velocity and size of each particle. Spatial density and density gradient with respect to time, angle and distance from Sun.
Electromagnetic radiation	Observe solar and planetary EM radiation.	Latitude dependent emission from solar surface, flares and planets if possible. Radio frequencies X and gamma rays.

The measurements of the interplanetary magnetic field, which apply only to the ecliptic plane, have shown the presence of fluctuations particularly in the direction of the field (Cahill 1963, Ness et al. 1964). It is generally agreed that the fluctuations are real and are related to the solar wind motion. The present situation is that the overall configuration of the interplanetary field cannot be explicitly expressed as a radial, spiral or a dipole type field. To elucidate this problem measurements are required over distances of a few AU and in a number of directions relative to the Sun. Furthermore there is now a need for measurements of the fine structure of the field fluctuations in such a way that they can be correlated with the behavior of the solar wind.

Extensive measurements have been made on galactic cosmic rays by monitoring the nucleonic component at the surface of the Earth and from balloons. Spacecraft data have confirmed the isotropy of cosmic rays in the region of the Earth orbit and verified the modulations in the flux (Forbush decrease (Coleman et al. 1961, Parker 1962, Gold 1962), 11 year solar cycle variations (Meyer and Simpson 1955, Parker 1958) and the 27 day recurring fluctuation (Goddard 1964b)) which are related to solar activity and solar flares. The solar flares expel energetic solar protons (or solar cosmic rays) but it is not quite clear how they travel through space to intercept the Earth's orbit, at first as a broad beam of radiation, and later as an isotropic flux (Meyer, Parker, Simpson 1956). Measurements of the energy and mass per nucleon and of the flux as functions of time and position over interplanetary space at all heliocentric latitudes and at distances certainly out to 5 AU will add considerably to the understanding of the propagation of cosmic rays and their interaction with the interplanetary magnetic field.

The spatial density of interplanetary matter is thought to be greatest in the ecliptic plane but the density gradient through space is not known (Elsasser 1963, Alexander, McCracken et al. 1962). Due to the Poynting-Robertson effect which causes dust particles to spiral into the Sun and solar radiation pressure which pushes the sub-micron particles out of the solar system, replenishing sources are required to maintain the observed distribution of particles in say the zodiacal light. Debris in the orbits of comets (Whipple 1958), and dust in the asteroid belt (Piotrowski 1953) have been suggested as possible replenishing sources. Data will be required not only on the particle flux and spatial density gradient but also on the velocity, size and direction of each individual particle encountered.

The electromagnetic radiation in space, which probably covers a range of 23 decades of wavelength (Goldberg and Dyer 1961) can be adequately observed from Earth orbital altitudes. However, it would be useful to measure from space the electromagnetic radiation that is latitude dependent, arising possibly in solar flares and other disturbances on the Sun, and from the planets.

Finally it is surely probable that electrostatic fields will exist in the solar system on a microscopic and probably a macroscopic scale. Very little knowledge exists of the electrostatic fields and severe measurement problems will arise in the detection of small electric field gradients.

3. BASIC EXPERIMENTS FOR INTERPLANETARY MISSIONS

The scientific objectives of interplanetary missions naturally determine the types and the sophistication of the required scientific instruments. The basic instruments required are:

a) Plasma probes
b) Magnetometers
c) Cosmic ray telescopes
d) Solar proton detectors
e) Solid particle detectors
f) Radiometers

and are summarized in Table 2.

Experiments involving all of the above basic types of instrument have already been flown, mainly in Earth orbital missions but also towards the moon and Venus (Cahill 1963, Sonnett 1963,Lust 1963, Kaiser 1963). It is therefore possible to specify both the weight and power requirements for these individual state of the art experiments with reasonable accuracy. However this is not an adequate criterion in that a very important aspect of interplanetary missions is the integration of the experiments in such a way that the data can be fully correlated when analyzed. This will require advanced

Table 2

BASIC EXPERIMENTAL PACKAGE FOR INTERPLANETARY MISSIONS

	Lbs.	Watts	Bits/sec	Remarks
Plasma Probes				
Energy spectrum	4	1	0.5	1 energy level/15 secs
Direction	6	5	4	Direction within ± 2.5 square degrees every 15 secs
Fast response	-	-	3	1 reading/msec, 2 secs/day, transmit 1 hr/day
Magnetometers				
3 axis fluxgate	5	1	1	1 measurement/15 secs
Fast response	-	-	10	1 reading/msec, 2 secs/day, transmit 1 hr/day
Rubidium vapor	6	8	0.25	1 measurement/2 mins.
Cosmic Ray Telescope				
10 meV - 1 beV	5	1	0.1	Measurement per cosmic ray intercepted
Solar Proton Detector				
100 keV - 10 meV	5	1	0.1	Measurement per solar proton intercepted. Storage required for solar events.
Ionization Chamber				
Neher - radiation dose	2	0.2	-	Integrates dose
Micrometeorite detector				
1-10 m^2 area	15	0.5	-	Measurement/collision
Radiometer				
200 mc/s - optional	5	1	0.1	Solar or planetary radio emission
	53	18.7	19.05	
Engineering data			2	
Approximate totals	55 lbs.	20 w	21 bits/sec	

plasma probes and magnetometers with short time resolutions and will involve particular attention being paid to the relative timing of each sample measurement. For example to measure the characteristics of small shock waves in the solar wind, and the associated magnetic field structure, will require measurements with a spatial resolution of at least an ion gyroradius, about 100 km, and perhaps even down to a Debye length of about 100 meters. This will require a resolution of the order of milliseconds. To measure the magnetic field in space, expressed in terms of axes based on the solar wind motion will require three axis magnetometer and plasma probe data in a time short compared to the period of fluctuations. It may thus be preferable say to take continuous and simultaneous magnetic field and solar wind data with good resolution over a period of several minutes but only once every few hours if the power demand is large. This may be more meaningful than uncorrelatable data from samples taken at regular short intervals.

Each of the basic experiments is discussed below.

3.1 Plasma Probes

The two basic types of plasma probe which have been used in spacecraft are the Faraday Cup and the curved plate electrostatic analyzer both of which were carried on IMP I. The differences between the two are largely mechanical since both use a modulated electrostatic field to accept plasma particles within a discrete energy band and both measure the flux of particles over a predetermined energy spectrum.

However, they have a disadvantage in that the output from the probes is a current proportional to the flux of charged particles being detected (protons and electrons must be measured separately) and the measurements are relative to the unknown charge on the spacecraft. Since the total solar wind flux at 1 AU over all energies only averages some 10^8 protons and electrons/cm^2/sec (Wheelock 1963), this represents a maximum possible current density of 1.6×10^{-11} amps/cm^2. To ensure that the solar wind can be detected even when the fluctuations minimize the flux and particularly when only narrow bands of the spectrum are being sampled, the tendency is toward large area, wide angle probes. This has meant that little detailed directional information has been obtained on the solar wind motion, and therefore little correlation has been possible between the instantaneous directions of the solar wind and the interplanetary magnetic field. To obtain this correlation, narrow angle detectors will be required which will have to be able to discriminate over the energy range of the solar wind and be arranged or be able to scan so that the predominant wind motion can be determined for each energy band. The loss in overall detection efficiency which accompanies narrow angle detection will probably mean that the present D.C. techniques must be excluded and that charged particle counters will have to be used to detect the individual protons and electrons, with direct pulse amplification being gained either in a gas or semiconductor or more probably using secondary emission in vacuum (Ogilvie et al. 1963).

It is suggested that two plasma probes be included in the basic experimental package, the first to sample the energy spectrum of the solar wind and the second to determine the principle motion of the solar wind plasma.

A curved plate electrostatic analyzer with a fairly wide angle of view (say 20° x 90°) would be suitable for measuring the energy spectrum of the solar wind. The effective cross sectional area can be made large enough to measure the spectrum at distances as far as 5 AU from the Sun (flux $\simeq 5 \times 10^6/\mathrm{cm}^2/\mathrm{sec}$) without too much difficulty. The weight of such a probe would be about 4 lbs. and it would require about 1 watt of power. To obtain a flux measurement to 5% in 20 energy ranges every 5 mins can be achieved with an average of 0.5 bits/sec of data.

The directional plasma probe should be capable of continuously monitoring the direction of maximum flux to within about $\pm$ 2.5 square degrees. A suitable flight tested instrument does not exist at present. A possible arrangement will consist of say five or preferably more collimated detectors which will scan and home in on the direction of maximum flux. Its duty cycle should be related to the magnetometer measuring the direction of the magnetic field and continuous measurements for 1 hour in 24 hours might well be adequate. The energy spectrum should also be measured with the first probe during the duty cycle so that the predominant energy band can be ascertained. The weight of this unit should be about 6 lbs. and it could require about 5 watts of power while in operation. The information rate to provide directional solar wind flux data every 15 secs to an accuracy of $\pm$ 2.5 square degrees should be about 4 bit/sec. However this only represents one use of this instrument. By comparing the output in the direction of maximum flux with that at say 90° to the maximum it should be possible to derive an ion temperature for the plasma. A sampling rate of 1 per minute should be adequate for this. Another important application is the use of the detector aligned with the principal plasma direction for short time resolution data. One reading per millisecond for two seconds each day will require about 10^4 bits of information to be transmitted. If this is transmitted over a one hour period the bit rate required will be some 3 bits/sec.

3.2 Magnetometers

The four basic types of magnetometer which have been carried in space probes are listed in Table 3 (Cahill 1963, Ness et al. 1964). The two devices most applicable to the measurement of interplanetary fields, which are generally less than 10γ (10^{-4} gauss) are the Rubidium vapor and the fluxgate magnetometers. The Rubidium vapor device measures the absolute value of the field, while the fluxgate has an adjustable range which requires calibration. The Rubidium vapor magnetometer is basically a scalar device, measuring magnetic field intensity and is only made three dimensional by the application of three orthogonal bias

Table 3

SUMMARY OF MAGNETOMETER TYPES

	Sensor/ Electronics Weight (lbs)	Power (watts)	Type of Measurement	Sensitivity	Range
Single search coil	0.5/1	0.6	Component perpendicular to spin axis	0.1γ	$0\text{-}1000\gamma$
Fluxgate saturable core (single range)	0.5/1	0.2	Magnitude in direction of core. Triaxial gives magnetic vector	0.1% of full scale or 0.25 gammas	Adjustable $\pm 50{,}000\gamma$ to $\pm 20\gamma$
Proton precession	0.1/3	144 watt-sec per measurement	Scalar - direction may be obtained by applying known magnetic fields	1γ	$10{,}000\gamma$-$60{,}000\gamma$
Rb vapor (two dual gas cell units)	5/10	8 incl. thermal active control	Scalar - direction may be obtained by applying known bias magnetic fields	0.01γ	3γ-$60{,}000\gamma$

fields in sequence. It is not possible to make measurements along three axes simultaneously with a single device and in the past a time of about 2 minutes has been necessary for a measurement of the magnetic vector. On the other hand with a fluxgate magnetometer it is possible to make simultaneous three axis measurements provided cross modulation between the axes can be avoided.

It is suggested that two magnetometers, one of each type be included in the experimental package to provide a contingency against failure and to make best use of the power supply and the overall range of the system.

A fluxgate magnetometer which will measure the magnetic vector in three components simultaneously is suggested for most of the measurements. It can be made within a weight limit of about 5 lbs. and will require an operating power of less than one watt. A suitable range would be say $\pm$ 25 gammas to allow for long term drifts and to cover the expected range of magnetic field intensity between 0.5 and 5 AU. The operating cycle can easily be adjusted to suit particular missions but continuous operation for one hour each day which gives a magnetic field measurement for each plasma energy level every 15 seconds to an accuracy of $\pm$ 2% will only require a transmission rate of 1 bit/sec. To provide short time response measurements from a triaxial fluxgate magnetometer will require some development because at present the gating pulses have a rate only of about 1000/sec. Nevertheless it is valid to think in terms of a three dimensional field measurement taken each millisecond which can be correlated with the solar wind data. If a three axis measurement is taken each millisecond for two seconds each day a total of 4×10^4 bits will be acquired. This can be transmitted over an hour at a bit rate of about 10 bits/sec. This transmission rate is quite large compared to the others in Table 2, but it can be accommodated on many interplanetary missions and it makes a new type of measurement possible.

The Rubidium vapor magnetometer is provided basically for calibration but also to serve if the magnetic field intensity exceeds the range of the fluxgate magnetometer. The weight of this device will be about 6 lbs. and the power demand of about 8 watts is largely consumed in maintaining the Rubidium lamp within fairly narrow temperature limits at about 40°C. A full three-dimensional measurement may take about 2 minutes by sampling each axis in turn in each direction. The information rate required will be about 0.25 bits/sec.

3.3 Cosmic Ray Detectors

3.3.1 Primary Cosmic Ray Telescope (10 meV to 1 beV)

Outside the atmosphere of the Earth it is possible to detect cosmic ray particles directly. The peak of the flux vs. energy distribution for primary cosmic rays occurs at about 1 beV and these particles can penetrate hundreds of grams/cm^2 of stopping materials. The technique used most effectively for determining the mass, range and energy of these high energy particles is to measure the rate of energy deposition ($\frac{dE}{dX}$) in a series of relatively thin detectors which may number more than six in a single device (Simpson et al. 1962, 1964). Because of the necessity that a detected particle should pass through the whole stack of detectors and because the diameter of the individual detectors is limited, the device will only give an output for particles in a narrow angle of acceptance. It hence becomes a telescope with defined directional properties. By calibration and comparison of each of the detector outputs both the mass and the total energy of the cosmic ray nuclei can be determined over a fairly wide range. It is also possible to measure the electron component of primary cosmic radiation. A telescope to measure total energy, range and $\frac{dE}{dX}$ for protons and alpha particles between 10 meV and 1 beV would weigh about 8 lbs. and require an operational power of about 1 watt. For continuous operation with limited storage available an information rate of 0.5 bits/sec should give a high counting efficiency and a resolution of about 5% per detector.

3.3.2 Solar Proton Detectors (100 keV to 10 meV)

It is possible for the above type of detector to be used for the low energy corpuscular emissions from the Sun (mainly protons) with energies that rarely exceed 1 beV. However they are unnecessarily complex for this purpose and simpler devices have been constructed. A thick scintillator with just one or at most two thin $\frac{dE}{dX}$ detectors in front of it would be adequate. The scintillation detector measures the total energy of individual particles and the $\frac{dE}{dX}$ detectors allow the instrument to be calibrated in terms of charge spectra and in terms of mass for singly charged particles. The total weight of the instrument will be in the region of 5 lbs. and will require a power of about 1 watt. The instrument should be sensitive to protons and alpha particles in the range 100 keV to 10 meV and the flux of these particles will be high during periods of solar activity but low between these periods. Thus there is a storage requirement for the data which should be such as to permit complete transmission at an average rate of 0.1 bits/sec.

3.3.3 Ionization Chamber

The experimental package should include an ionization chamber of the Neher type which will effectively integrate the total ionization produced in a known volume of standard density air and thus be a radiation monitor. The instrument can be arranged to give an output pulse after a predetermined amount of ionization has occurred or at regular time intervals. It will reset after each pulse. It will be a simple reliable instrument weighing only 2 lbs. and requiring an average power of about 0.2 watts. The data rate from this radiation monitor will be negligible except perhaps during a large solar flare.

3.4 Micrometeorite Detectors

The masses of the particles in interplanetary space range from about 10^{-15} gms to many tens of grams. The flux of particles increases rapidly as the mass decreases as can be seen in Figure 1. In order to anticipate a significant number of collisions with a micrometeorite detector of reasonable area ($\simeq 1\ m^2$), it is important that particles of low mass ($\simeq 10^{-6}$ gms) be detected. The required information from micrometeorite detectors, and this has not been completely accomplished yet, is to determine all the dynamical properties of each impacting particle and the total number of collisions per unit time and area. The dynamical properties required are size, mass, density, velocity, energy and momentum some of which are implicit.

The most widely used type of detector up to the present time has been the acoustic device which records the impact of the particle collision. The signal can be interpreted in terms of the momentum of the colliding particle. The impacting energy of a particle can be detected using a crystal detector in which the light output from the collision with the crystal is the signal. Another type of detector, which has not been flight tested but which seems promising uses two light screens. The scattered light as a particle passes through the screens is detected, and by the time separation, the velocity of the particle can be calculated (Neuman 1963). A virtue of this detector is that it does not destroy the particle in the measurement.

The methods which are now rapidly gaining usage rely on the penetration of specific materials by the particle. Simple devices include a matrix of pressure cans or wire grids,

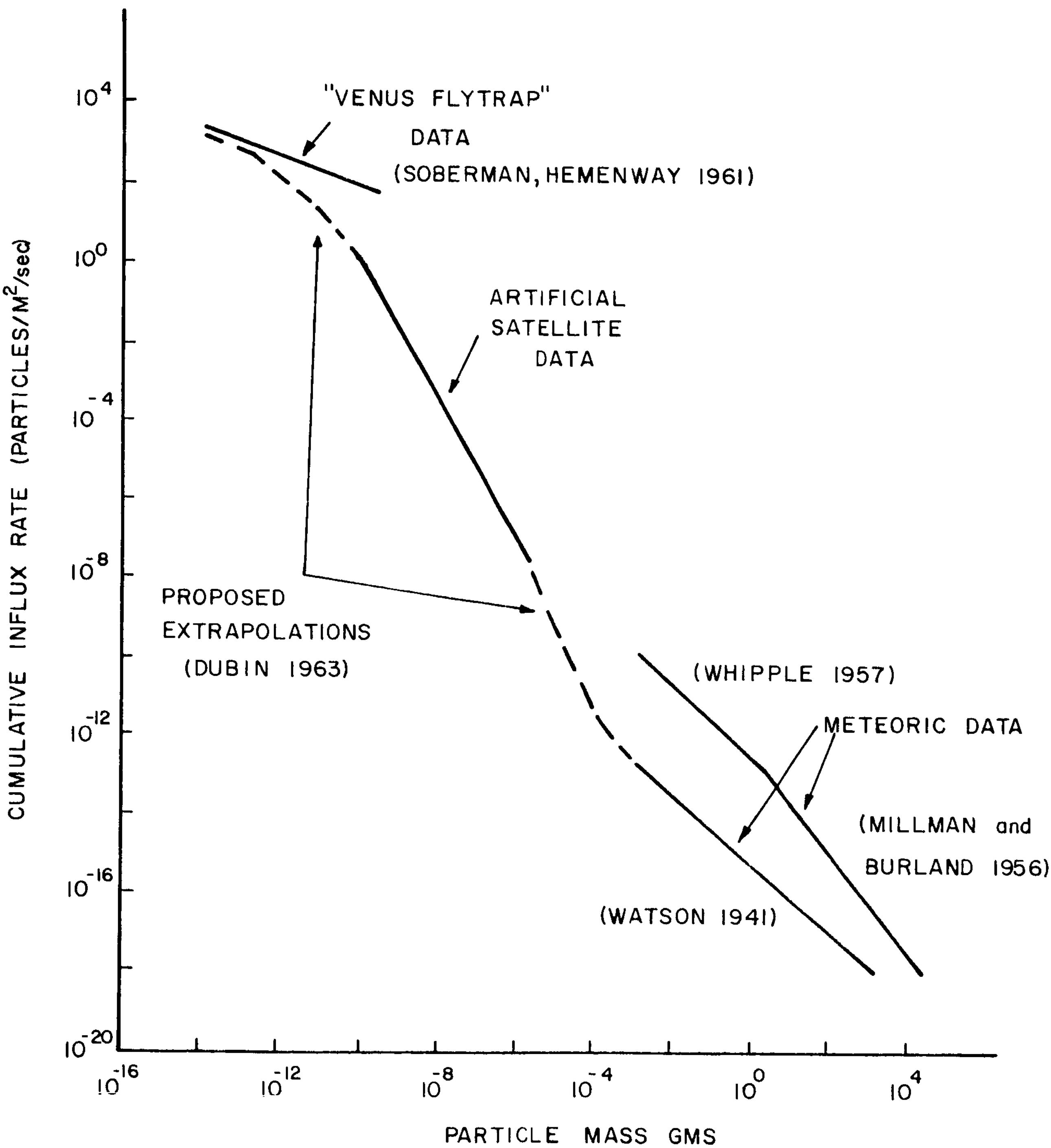

FIG. I. NEAR EARTH PARTICLE FLUX VERSUS MASS the information being that cans have been punctured or wires broken by impacts and hence are not included subsequently in the effective area of the detector. More complex devices optically determine the size and shape of penetration holes in thin plastic films and count the collisions by capactor action between conducting films on each side of the plastic. This device usually without the optical system is becoming popular for large area detectors largely because of the small weight/unit area, low power consumption, compact folded volume, and high counting efficiency.

It is suggested that the latter type of film capacitance detector be included in the experimental payload. Using the Mariner results of one count in 150 days as a guide it would seem that a large area device is required for interplanetary missions say at least 1 square meter. This may be inadequate for a statistically significant number of collisions in non-ecliptic regions and an area as large as 10 square meters may be required. However in comet orbits and in the asteroid belt it is probable that the collision rate may be quite high (1 count/min was obtained as Mars I* traversed the Taurid meteor stream) and it will therefore be necessary to retract a significant area of detector to preserve its effective cross section for later parts of the mission. The remaining small area should be such that it will still give sufficient data on regions of high particle density.

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* Soviet Mars probe.

For a 10 m^2 film the weight will be only one or two lbs. and the associated electronics would weigh a further 2 lbs. Adding to this the structure to deploy and retract the detectors will give a total weight of say 15 lbs. inclusive. The average power requirement will be less than 0.5 watts and the bit rate will be negligible except in regions of high particle density.

3.5 Radiometers (optional)

For certain interplanetary missions it will be advantageous to include a radiometer in the experimental package. The advantage to be gained compared to Earth measurements lies largely in the relative position of the spacecraft with respect to the radio source rather than in increased resolution or signal strength. For radio emission from the Sun and particularly from solar flares,say 200 mc/sec, non-ecliptic measurements might help in determining the directionality of the radio emission. In the case of planets, and especially Jupiter, decameter radiation measurements from missions which fly by the planet may be useful in interpreting the measurements taken from the Earth. Neither a narrow beam width nor high resolution is required but simply a knowledge of the radiation in the source-spacecraft direction. Therefore a dipole array will probably be adequate as an antenna. The weight of the system should be only about 5 lbs. and a power of 1 watt should be adequate. The information to be transmitted is the relative intensity of the signal and should require only a nominal bit rate less than 0.1 bits/sec.

There are many radio frequency experiments which should and could be carried out from interplanetary spacecraft. In the interest of keeping the basic spacecraft weight to a minimum, consideration has only been given to those which take advantage of the different perspective of solar system sources obtained from deep space trajectories. The two suggestions above are used as examples.

4. FLIGHT PARAMETERS FOR INTERPLANETARY SPACE MISSIONS

The trajectory is an important aspect of all interplanetary space missions, both in and out of the ecliptic plane, because the principal objective is collection of data along the flight path. It is required that the optimum trajectory be selected in terms of its suitability to the scientific objectives, its compatibility with launch vehicles, spacecraft weight, reliability and communications demands. Therefore the flight parameters have been determined after consideration not only of the ideal velocity and time of flight parameters but by selection from families of minimum energy deep space trajectories to heliocentric distances between 0.5 and 5 AU and heliocentric latitudes between $\pm$ 50°. The families of trajectories are given in Appendix 1 at the back of this report.

There are, in fact, a large number of trajectories to a given point in space and each will be associated with a specific time of flight, ideal velocity and orbit plane. When an interplanetary mission has a target area specified by a heliocentric latitude only, then the absolute minimum ideal velocity flight to the given latitude is possible. If in addition, the target area is specified by heliocentric distance as well as latitude, then use can be made of the minimum ideal velocity flight to the required r and β, and there will be no launch window constraints.

Figure 2 shows the minimum ideal velocity required to reach a given heliocentric distance and latitude and also shows the required time of flight. It can be seen that, for the flight requiring an absolute, minimum ideal velocity to a given latitude, the heliocentric distance at intercept will be less than 1 AU, i.e., inside the Earth's orbit. This same information is provided in more graphical form using the accessible regions plotting technique of Figure 3 (Narin 1964). In this figure, the contours show the regions in the solar system which can be reached with a given ideal velocity with no account taken of longitude.

Figures 2 and 3 are very useful for gaining a first idea of the flight parameters but do not provide sufficient information for a mission study. Fortunately there is an intermediate step for deducing the flight path data without having to compute detailed flight parameters for each suggested mission or trajectory. This takes advantage of the fact that when no specific target body and target ephemeris is involved a family of minimum energy trajectories can be drawn for flights to various heliocentric distances each at a given latitude. A series of such families of trajectories are given in Appendix 1

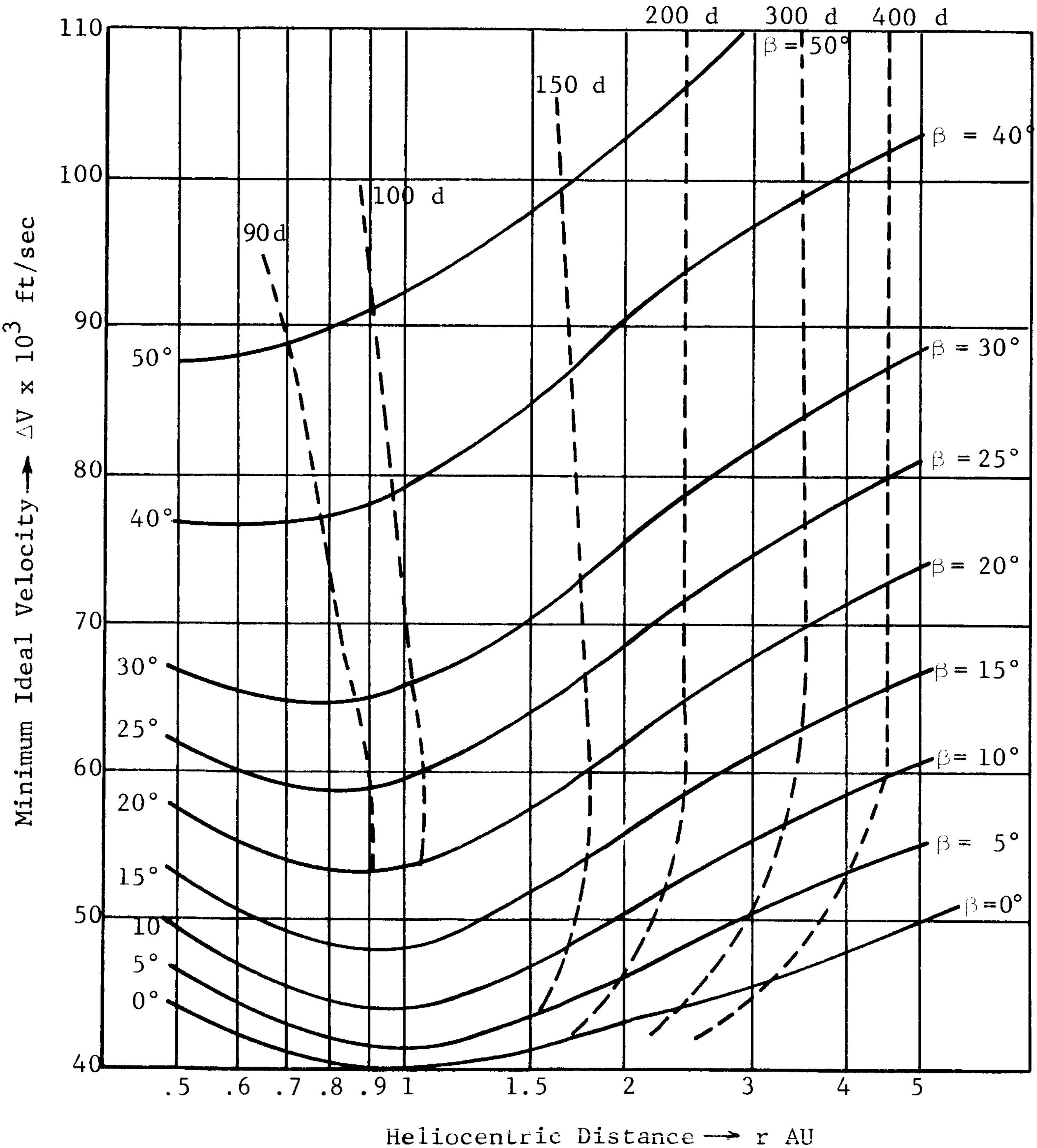

Fig. 2 IDEAL VELOCITY AND TIME OF FLIGHT FOR MINIMUM ENERGY TRAJECTORIES TO (r,β)

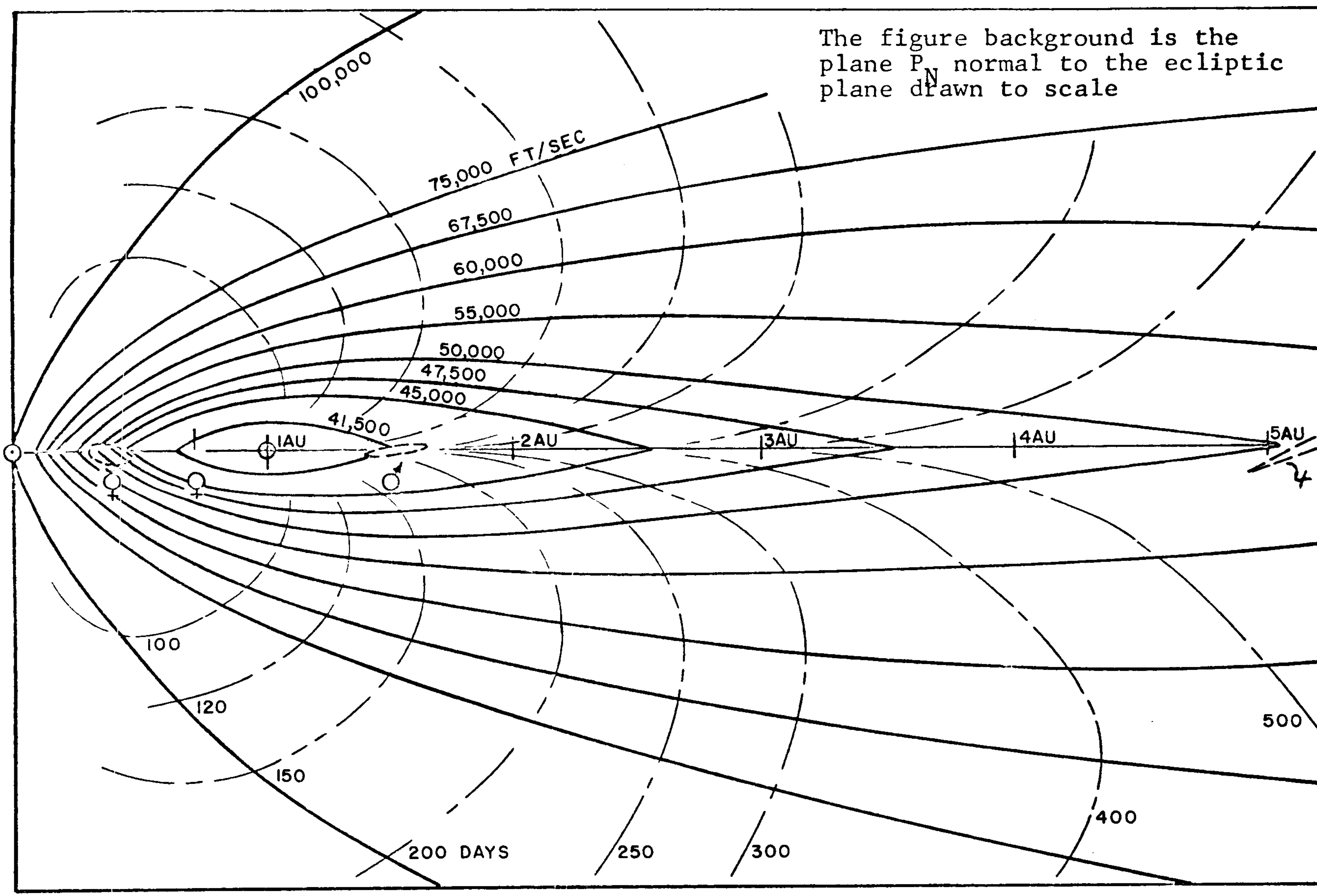

Fig. 3 IDEAL VELOCITY CONTOURS FOR MINIMUM ENERGY BALLISTIC TRAJECTORIES

for target latitudes between 0° and 50°. Examples are shown in Figures 4 and 5.

Figures 4 and 5 show minimum energy trajectories to latitudes of $\beta = 20°$ each being drawn as it would appear in the orbit plane. It is important to note that the orbit plane will generally have an inclination greater than β and therefore the spacecraft will travel above the required latitude for some portion of its orbit. This is shown by the full-line part of the trajectory. The locus of intercept points at $\beta = 20°$ is shown for these minimum energy trajectories and hence other trajectories can be easily interpolated between those shown. Time of flight contours have been included so that the relative positions of the Earth and the spacecraft can be seen along the flight paths.

The families of trajectories have further use in the construction of approximate multiple minimum energy flight paths. Figure 6 shows, as an example, three trajectories in the ecliptic plane which will place three spacecraft in line with the Sun after 500 days from the first launch. The heliocentric distances are 3 AU, 4 AU and 5 AU respectively.

The use of the generalized trajectories enables a good approximation to be obtained for the time of flight, minimum ideal velocity, communications distance, time above the desired latitude, heliocentric flight angle and the inclination of the orbit plane. It is only after trajectory selection has been

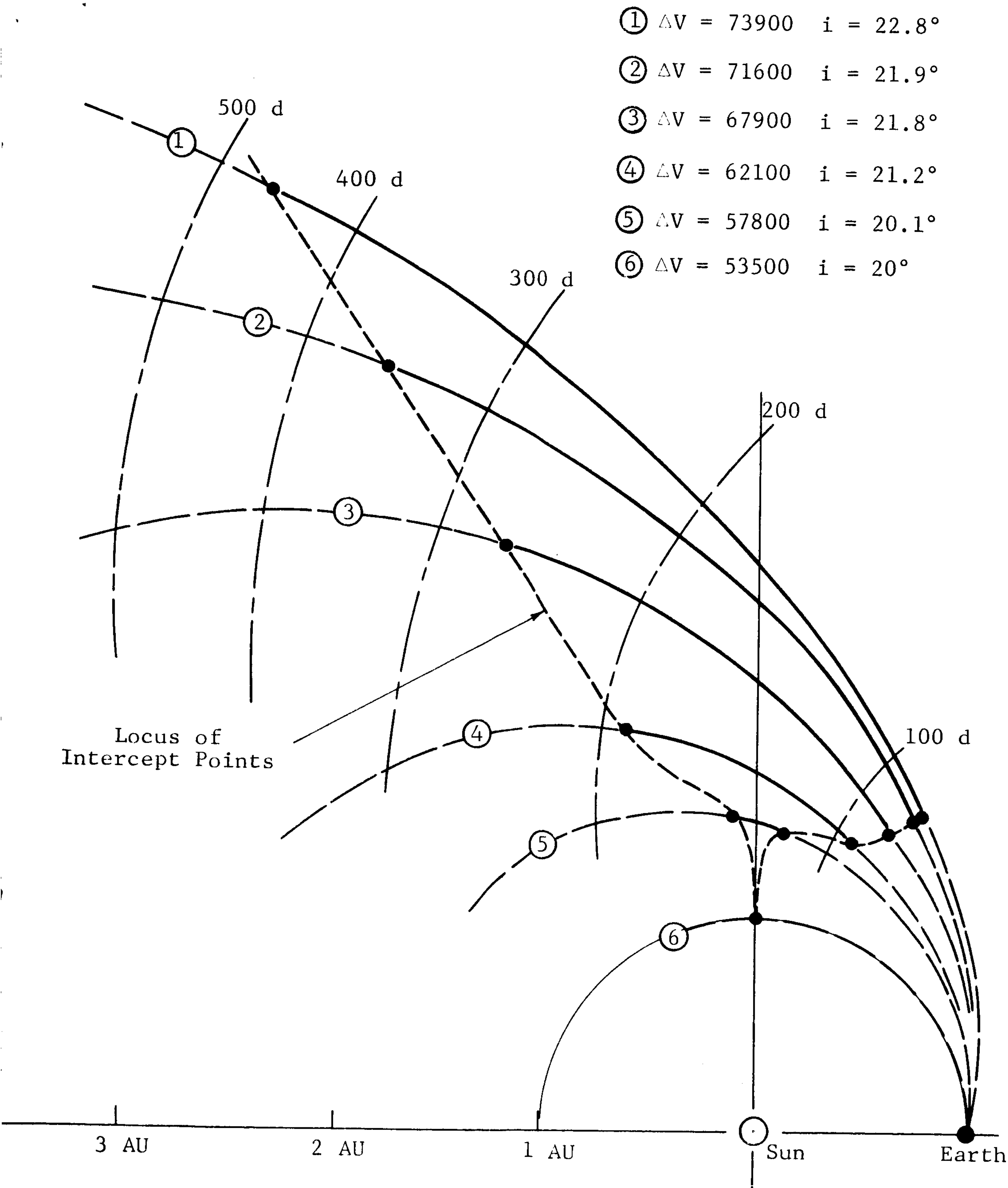

Fig. 4 FAMILY OF MINIMUM ENERGY TRAJECTORIES TO A LATITUDE OF 20° (1 < r < 5 AU) (TRAJECTORIES DRAWN IN PLANE OF ORBIT AT INCLINATION i°)

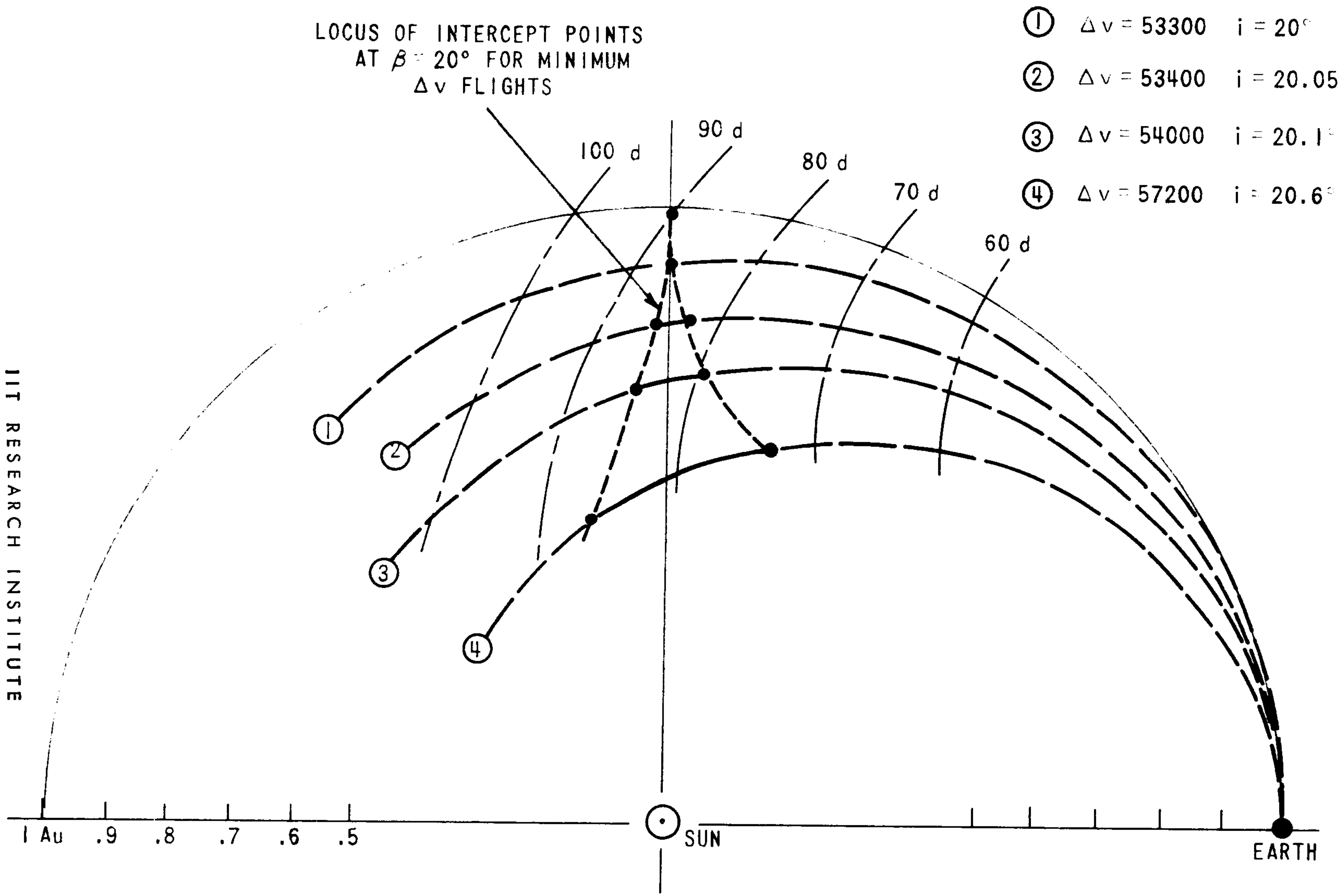

FIG. 5 FAMILY OF MINIMUM ENERGY TRAJECTORIES TO A LATITUDE OF 20° (0-5 < r < 1 Au).
(TRAJECTORIES DRAWN IN PLANES OF ORBIT AT INCLINATION i°).

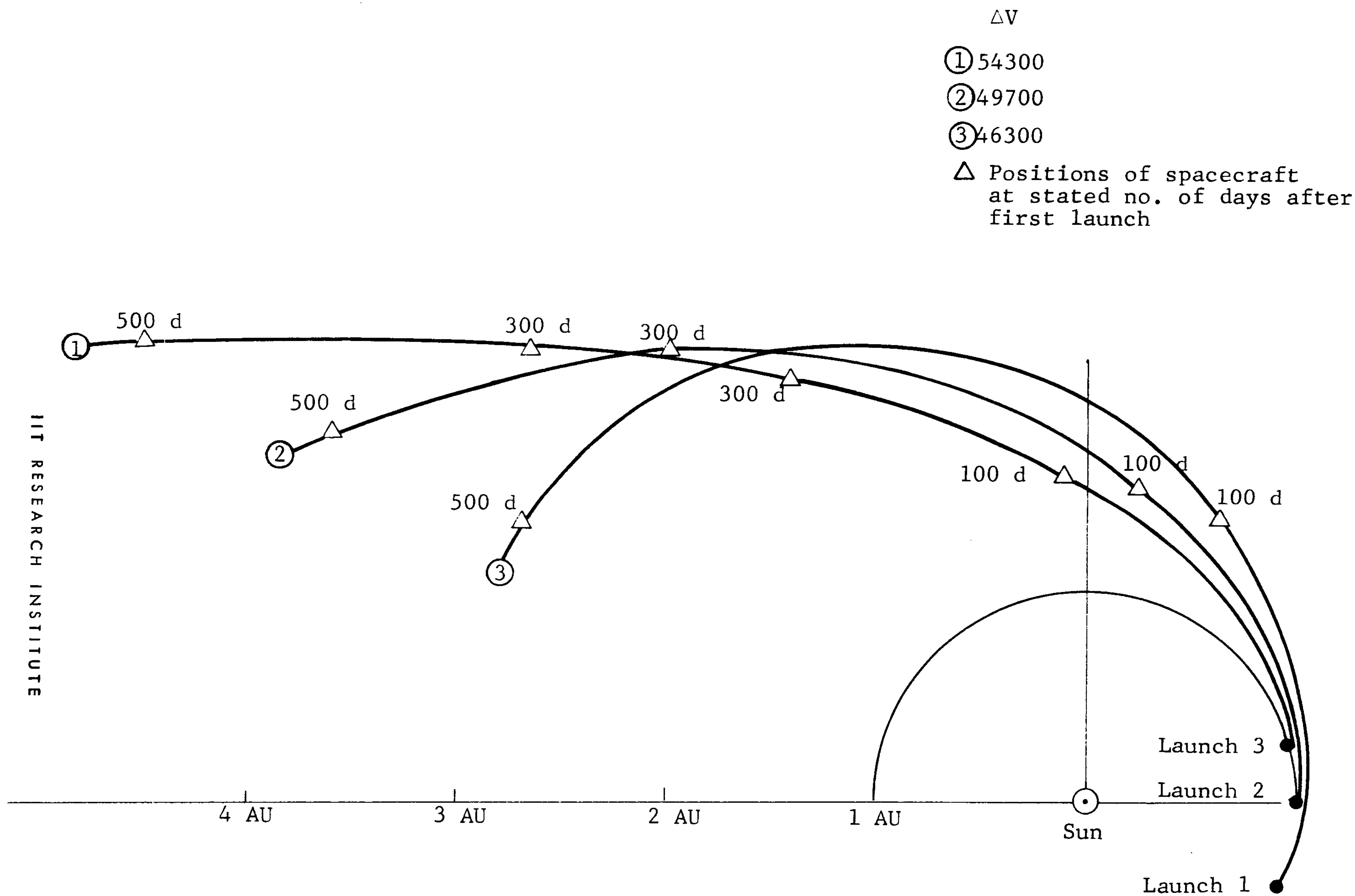

Fig. 6 EXAMPLE OF MULTIPLE MINIMUM ENERGY TRAJECTORIES IN ECLIPTIC PLANE (ALIGNMENT WITH SUN AFTER 500 DAYS AT 3 AU, 4 AU and 5 AU)

accomplished on this basis that any detailed trajectory parameters need be specifically computed.

5. MISSION CONSTRAINTS

An interplanetary space mission at first sight would seem to have very few mission constraints and this may be particularly true of the earliest missions. For these missions the trajectory may if necessary be corrected shortly after launch and thereafter the spacecraft would simply coast through space. Despite this apparent simplicity, all of the normal mission constraints have to be considered in the selection of the trajectory and in the maintenance of the spacecraft in an operational condition. The mission boundaries can be divided into classes:

1. Functional Constraints

   Trajectory selection
   Launch window
   Midcourse guidance
   Attitude control
   Communications and telemetry
   Power Supply
   Reliability

2. Experimental Constraints

   Spatial resolution
   Data correlation

3. Environmental Constraints

   Meteoroid damage
   Radiation damage
   Temperature

5.1 Functional Constraints

The major constraint on the function of interplanetary missions will be associated with the selection of the trajectory. For most missions there will be no launch window constraint and no need for midcourse guidance. The attitude control system can be made to operate in a set of reference axes commonly defined for all missions irrespective of the latitude or distance of the spacecraft. The communications system, the power supply and the anticipated reliability for the missions will be independent of the inclination of the orbit plane but critically dependent on the distance of the spacecraft.

5.1.1 Trajectory Selection

The first selection must be between ecliptic and non-ecliptic trajectories and if non-ecliptic with what relationship between heliocentric distance, latitude and time. The regions of space traversed by the spacecraft and the duration of the stay near particular latitudes or distances are selected basically on the grounds of scientific utility but must be complemented in terms of the ideal velocity, time of flight and communications distance. In particular the ideal velocity requirement must be used with present or predicted launch vehicle performance characteristics to determine the payload weight which can be injected. The time of flight reflects mainly on the reliability prediction, and the communications distance on the weight of the telemetry system. The families of minimum energy trajectories given in Appendix 1 are useful in determining the best compromise between the demands of many experiments and, being minimum energy, ensure that the maximum injected payload will be possible for the given mission. The final stage in selecting the trajectory arises because of the relatively fixed weight of the experimental payload. The selected trajectory is adjusted in latitude or distance so that the appropriate vehicle performance is fully utilized in injecting the spacecraft. In the examples given later the mission to 13.5° latitude was reduced from a 15° mission to make it compatible with the specific launch vehicle.

5.1.2 Launch Window

Interplanetary missions in general have target areas which are not specified in heliocentric longitude. This does not necessarily imply that the solar system is symmetrical but that, with the present limited knowledge, it is not possible to predict the special significance of any interplanetary longitude far enough in advance of a mission. In this event, there will be no launch window constraints on interplanetary missions, and the minimum energy trajectory will always be possible. This situation will clearly be modified if planetary missions are combined with interplanetary ones, and then the launch window is determined by the planet's ephemeris.

5.1.3 Midcourse Guidance

The interplanetary missions studied here are largely exploratory and are aimed at different regions of space to see if continuity or significant differences exist as functions of latitude and distance. Therefore it is not possible to select a specific target area and demand that the spacecraft must pass through it. This latter situation will however probably arise as a result of these initial exploratory missions. Until this time, the spacecraft will require correction only for the initial launch errors and then be allowed to coast through space without further midcourse correction. The maximum launch error may be assumed to be 10 m/sec in each of three orthogonal reference directions with each component statistically uncorrelated. Thus a maximum correction of only about 17 m/sec may be required, if it can be applied within a few days after injection.

5.1.4 Attitude Control

It would seem that all interplanetary missions in the foreseeable future will require attitude control. Those missions going to greater distances than 1 AU from the Earth will require directional communications antennas which must be oriented towards the Earth. Those remaining close to the Earth may use an omnidirectional antenna but will also probably be powered from solar cells which must be oriented towards the Sun. For all missions the one plane which will remain heliocentrically fixed will be the orbit plane which will contain both the Sun and the spacecraft. Thus a suitable static reference frame will have one axis pointing towards the Sun, one axis perpendicular to the orbit plane and the third axis mutually orthogonal and in the plane of orbit. This reference frame will be convenient for interpreting the data heliocentrically and for aligning solar panels but it should be remembered that the Earth does not move in the orbit plane except for ecliptic missions. This will mean that the antenna and certain of the directional instruments may have to be supplied with individual scanning drives or movable collimators.

5.1.5 Communications and Telemetry

The communications equipment for interplanetary missions is governed mostly by the bit rate and the distance over which the information must be transmitted. Figure 7 shows a graph of idealized transmitter output power as a function of communications distance and information rate. The graph is idealized only insofar as -3 db has been allowed for performance losses. A state of the art figure is more like -10 db. The spacecraft transmitting antenna is assumed to be a 3 ft. diameter (25 db) dish and the receiving antenna either the 210 ft. DSIF dishes or more probably the 85 ft. dishes. The raw power supplied to the transmitter may be up to 10 times the transmitted power.

The communications distance only increases slightly as the inclination of the spacecraft orbit is increased and it is not unreasonable to ignore heliocentric latitude effects in the first estimation of the transmitter requirements.

For transmission over large communications distances the penalty is paid in the weight and power demand of the transmitter and the weight of the antenna. If it is assumed for interplanetary missions that the amount of experimental data

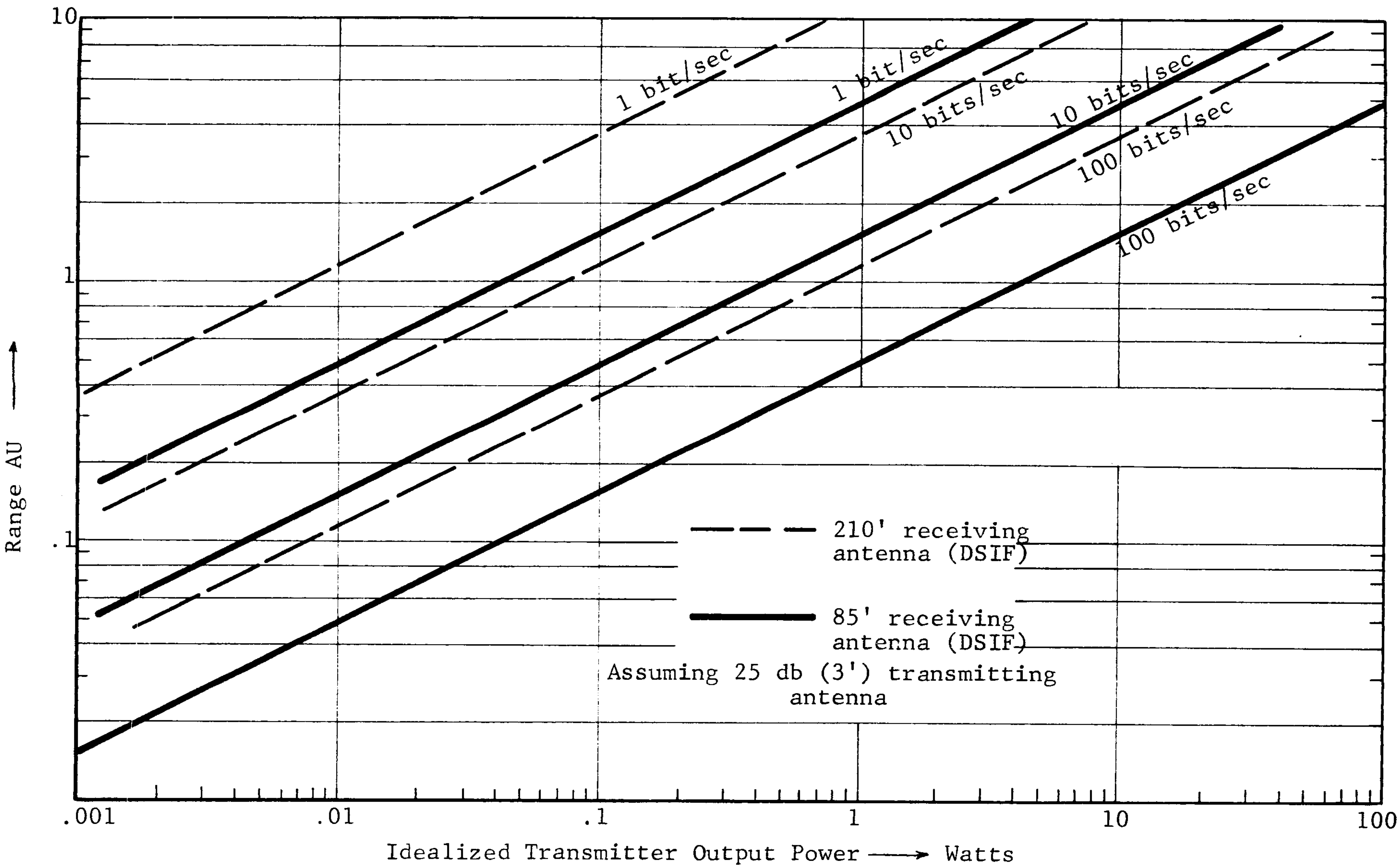

Fig. 7 TRANSMITTER POWER (WATTS) VS. RANGE AND INFORMATION RATE

acquired is about the same from day to day, and this will probably be true, then the transmission information rate is essentially fixed and minimized if transmitted at a steady rate over each 24 hours. However it may not be desirable to receive the data on Earth for a full 24 hours each day because of the commitment to the receiving stations. Two alternatives are possible. Either the data can be stored on board and transmitted at a fairly high rate over a period of say one hour each day, or the data can be transmitted continuously and economically but only be sampled on Earth for, say, an hour every 4 hours.

This last alternative has advantages in that it permits large irregular zones in space to be detected and it minimizes the transmission and data storage on the spacecraft. The only data which cannot be relayed in approximately real time are from the short time response plasma probe and magnetometer. These will require a storage capacity of about 5×10^4 bits.

The computing and sequencing equipment on board will also be minimal and will be largely the same for all interplanetary missions. The data to be transmitted should only be conditioned before it is sent and the data reduction should take place on Earth. This is particularly pertinent if only 25 percent of the data is to be received anyhow.

5.1.6 Power Supply

The type of power supply will be determined mainly by the transmitter requirements which will be a function of the heliocentric distance to be attained by the spacecraft, and only secondarily by the experimental payload which will not vary much from mission to mission.

Thus for missions on which the spacecraft remains within say 2 AU from the Sun and the Earth, a solar cell power supply will be adequate irrespective of the inclination of the orbit. A currently attainable weight of 0.75 lbs/watt is assumed for solar panels. For missions to larger distances, a radioactive source will probably be required and which would be isotopic for power levels up to say 250 watts (electrical) and a SNAP type reactor system for larger demands. Specific weights of about 1 lb/watt are assumed for radioactive power supplies exclusive of radiation shielding. The shielding should be such that it not only protects the spacecraft from radiation damage but also prevents the radiation from affecting the interplanetary particles and fields in the vicinity of the spacecraft.

5.1.7 Reliability

The anticipated reliability of a mission can be reduced to an estimation of how long the spacecraft may be assumed to be operational. In fact one really requires a high probability that all the experiments and equipment will be still functioning in the target region. Although specific hazards exist at particular parts of the mission, i.e., high acceleration at launch or high radiation levels passing through the Van Allen belts, the major criterion is the flight time to the target area. The time of flight is naturally one of the basic flight parameters considered in selecting the trajectory and it can be seen that it is not influenced much by the latitude of the target area (Appendix 1). As a first approximation, reliability can be considered as independent of the inclination of the flight plane. Nevertheless it is difficult to predict future reliability coefficients which are meaningful and account for advances and sophistications in measurements and system techniques. As a poor best, approximate figures for state of the art system lifetimes are used insofar as a useful life of nine months is assumed practical now, eighteen months in the latter part of the decade and greater than two years beyond 1970.

5.2 Experimental Constraints

One of the experimental problems in investigating the interplanetary medium is to determine how typical are the measurements when only sample data are taken at consecutive times and at consecutive points in space by a single spacecraft. This is particularly pertinent when the parameters being measured exhibit considerable fluctuations as has been found for ecliptic interplanetary measurements of the solar wind and magnetic field.

Three basic possibilities exist: a) the irregularities are essentially stationary, b) they are being propagated through space and reflect the conditions at their source, c) there is a basic steady state configuration with superimposed fluctuations. It may be difficult to resolve between these three conditions. The problem is really one of resolving between say a large scale region of turbulence moving with a high velocity and a small local turbulence, which is almost stationary in space. With Earth orbital missions, this problem is resolved by repeated passes of the same zone which allows recognition of moving and stationary phenomena. For interplanetary missions where only a single pass is anticipated this solution is not possible. However one answer lies in the use of two or more spacecraft separated in space to observe stationary and moving fronts through time correlation.

It has become obvious that there are interrelationships and dependencies between the particles and fields in interplanetary space. The most obvious ones are between the solar wind and the magnetic field and between solar flare emissions and the galactic cosmic ray flux. A certain mission restraint is that not only should these parameters be measured but that the data should be obtained in such a way that correlation will be possible. In fact it is conceivable that relatively small samples of data which are correlatable would be far more indicative than continuous but insular data.

5.3 Environmental Constraints

The hazards of the space environment to spacecraft, and the weight and complexity of the shielding required to minimize these hazards are both position and flight time dependent. Therefore the environmental constraints must be considered in the light of the selected trajectory. The design of meteoroid shields is presently based on extrapolated empirical data. However from the anticipated flux of particles in interplanetary space it is not expected that damage will be a major criterion except in the asteroid belt or possibly while crossing comet orbits. For interplanetary flights of a duration from six months to a year, the question of meteoroid shielding could probably be settled arbitrarily in terms of the probability of collision, the value of one particular spacecraft performing for a long period compared to the total number to be launched, and the importance of data obtained late in a trajectory as opposed to the earlier parts.

A similar situation exists regarding radiation damage from both internal and external sources. External sources are principally the solar wind, cosmic rays and perhaps to a small extent the electromagnetic radiation from the Sun. Internal sources are isotopic or nuclear reactor power sources and in general these must be shielded due to their proximity to the spacecraft. However shadow shielding is a useful weight saver for internal radiation shielding in that only line of sight radiation to the susceptible areas of the spacecraft is shielded against.

The black body temperature in interplanetary space is a function of heliocentric distance. In practice this is not a major concern since most of the spacecraft can be made to operate over quite wide temperature limits. The individual exceptions to this can be isolated and operated in a relatively small temperature controlled environment either through selective thermal shielding or through electrical ovens.

6. MISSION OUTLINES

Interplanetary missions must naturally take their place alongside all solar system missions. They have been considered as a separate series here because of the special advantages offered by freedom from launch window restraints, by minimal guidance and control demands, by the basic similarity in the payloads and spacecraft and in some cases by their feasibility in the near future. These reasons however only support the scientific importance of understanding the nature of and the interrelationships between particles and fields in space and how they then influence the environments of all bodies in the solar system. The limited knowledge of interplanetary space means that the missions must be considered as exploratory and has made it necessary to consider non-ecliptic missions as well as ecliptic ones. The heliocentric latitude and distance (and perhaps the longitude) of each measurement may have an important bearing on the interpretation of the data. It must be stated that although a series of exploratory missions have been derived here an essential ingredient of an interplanetary program must be the recognition of new and unexpected configurations in space which may warrant further detailed study and perhaps specific missions.

The region of space studied in most detail is bounded by latitudes of $\pm$ 50° and lies between radii of 0.5 and 5 AU. Within these limitations four mission profiles have been developed as worthy of fulfillment. However they are by no means exclusive, and may even be considered more as samples. For instance, it may be possible to support a purely interplanetary mission with an, as yet, unspecified Earth orbital mission to provide two measuring points in space. Similarly an interplanetary mission may provide useful support to future planetary investigations by simultaneously monitoring space beyond the planet's environment. Such combinations have not been described but could be quite quickly deduced from the trajectories presented in the appendix.

The missions outlined fall into four broad categories: 1) minimum energy ecliptic, 2) minimum energy to a given latitude, 3) minimum energy to a specific latitude and distance, and 4) multiple missions. They are summarized in Table 4.

The launch vehicles required for most of the missions have to be fairly advanced in view of the high ideal velocities. The seven basic launch vehicles with their approximate performances are specified in the mission profiles. These are:

Table 4

SUMMARY OF MISSION PROFILES

Mission	Target Area Dist. r AU	Target Area Lat. β°	Time of Flight TF days	Ideal Velocity ΔV ft/sec	Inc. of Orbit i°	Comm. Dist. RC AU	Payload Weight lbs.	Possible Launch Vehicle
Ecliptic to 5.2 AU	5.2	0	500	55,000	0	4.2	360	Atlas Agena + H.E.
Absolute Min. ΔV to $\beta = 13.5°$ Lat.	0.9	13.5	90	47,000	13.5	0.25	240	Floxed Atlas Centaur
Absolute Min. ΔV to $\beta = 22°$ Lat.	0.85	22	90	56,000	22,01	0.5	260	Atlas Agena + H.E.
Min. ΔV to r = 3 AU, $\beta = 15°$	3	15	265	61,500	17.2	4	350	S1B Centaur + L.E.

H.E. = High Energy Stage (I_{SP} = 455 secs)

L.E. = Low Energy Stage (I_{SP} = 300 secs)

1 Thor-Agena D = 600 lbs. to 100 N. mile orbit.
2 Atlas-Agena = 700 lbs. to escape
3 Floxed Atlas-Agena = 1300 lbs. to escape
4 Atlas-Centaur = 2100 lbs. to escape
5 Floxed Atlas-Centaur = 2900 lbs. to escape
6 Titan III-C = 5400 lbs. to escape
7 Saturn 1B-Centaur = 13,000 lbs. to escape.

In most cases one of the following kick stages is required

LE = kick stage with I_{SP} = 300 secs

HE = kick stage with I_{SP} = 455 secs.

It is assumed that the basic launch vehicle places the spacecraft and additional stages into a 300 N. mile parking orbit. The Saturn V launch vehicle has not been thought applicable for small payloads, since three or more additional stages would be needed to make full use of its potential. A possible alternative to a multi-staged Saturn V is provided by a semi-continuous low thrust stage launched with a Saturn 1B launch vehicle (JPL, 1963). This type of system appears to offer considerable advantages for large deviations from the ecliptic plane. However since the availability of low thrust stages is not well enough known at present, a mission based on their use has not been included.

A payload breakdown is included for each of the missions described. The same basic experimental package is included for each but the power and transmitter demands are specific for each mission. Three payload weights are presented. The first represents the payload weight in the target area after the launch correction and attitude control propellants have been used up. The second is the weight at the start of the trajectory but after injection and after the injection stage and its adapter have been jettisoned. Finally the total effective weight of the spacecraft is given and includes weight allowance for the spacecraft shroud and the injection stage adapter. The allowance takes into account the fact that the shroud is released before injection and therefore only acts as payload for part of the total launch vehicle thrust time. The effective weight allowance is given for each mission. The total effective weight is the one which has been used with the launch vehicle performance curves to give the launch vehicle-payload combinations listed.

6.1 Minimum Energy Ecliptic Missions

Many missions have successfully collected interplanetary data from space near the Earth's orbit. These include the Explorer series of Earth orbital missions and particularly IMP 1. Valuable data has also been obtained from Pioneer flights and the Mariner Venus mission. These series are being actively continued with the impending IMP, Advanced Pioneer and Mariner Mars missions. Therefore the first ecliptic mission outlined here marks a new phase of interplanetary exploration in the ecliptic plane. It is a mission to the region of Jupiter's orbit (5.2 AU) and with a flight time restricted to 500 days. The launch can be phased in with an approach to Jupiter but without introducing launch window constraints. It may be possible to pass within 0.25 AU of the planet and still to detect the supposedly very large magnetic field, if the spacecraft is in the magnetospheric tail, and to monitor the radio emission from the planet and its environment. Figure 8 shows the trajectory for this mission. The position of the Earth has been marked to correspond with the flight time markings on the trajectory, and the approximate relative position at launch, of Jupiter has been marked. At about 300 days after launch the spacecraft will be approximately in opposition with the Earth and communications will be difficult for about a month. Table 5 shows the breakdown of 360 lb. payload and the mission profile using an Atlas Agena launch vehicle and a high energy additional stage ($I_{SP} = 455$).

Additional ecliptic missions will be required in the long range exploration of the solar system including missions to specific regions of interest which will probably be revealed by this 5 AU probe and to as far out as the boundary of the solar system. The latter missions may require high ideal velocities up to and beyond 100,000 ft/sec for reasonable flight times and have not been detailed here.

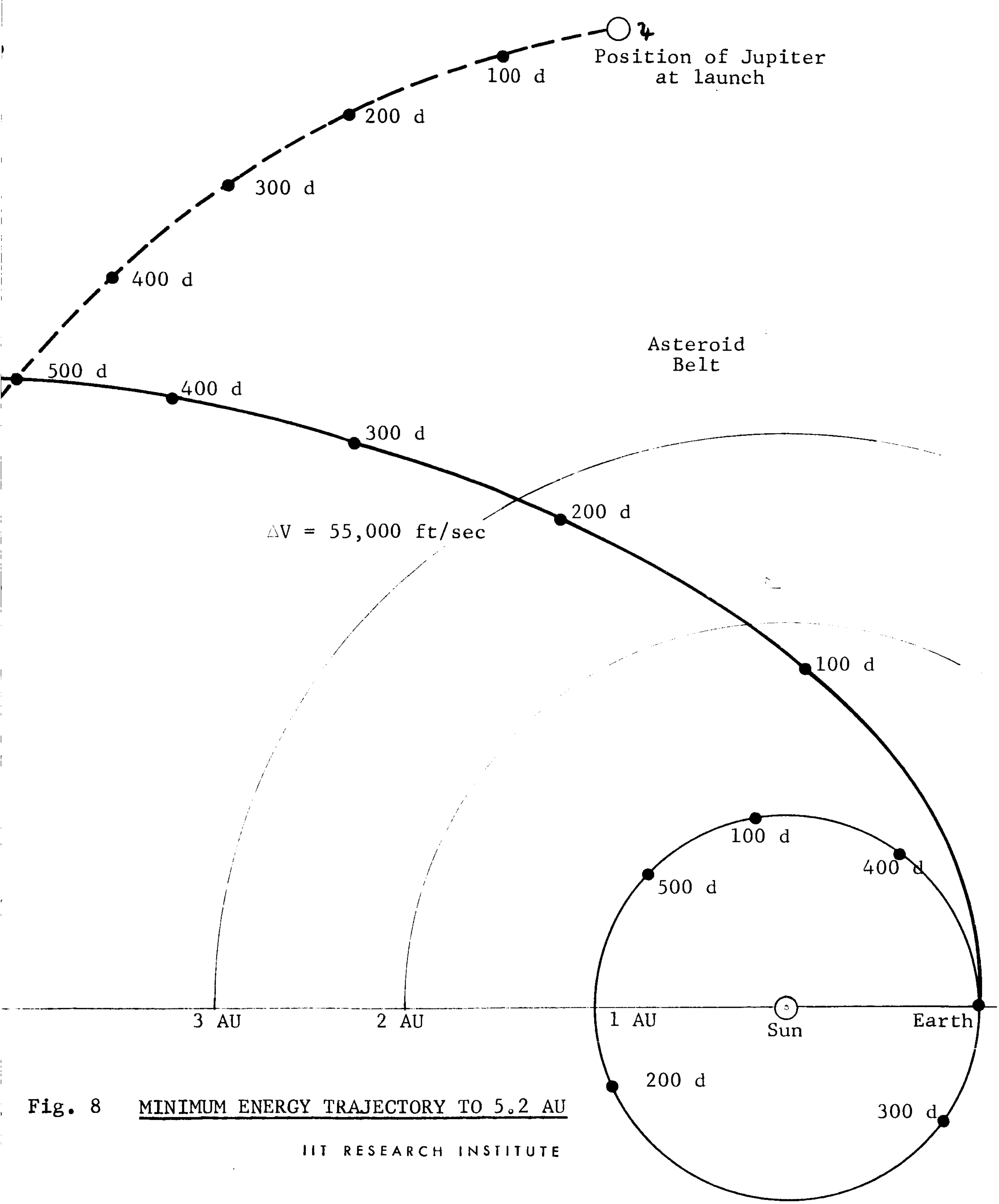

Fig. 8 MINIMUM ENERGY TRAJECTORY TO 5.2 AU

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Table 5

INTERPLANETARY ECLIPTIC MISSION TO 5.2 AU
(500 day limit)

Trajectory Parameters

Ideal Velocity ΔV = 55,000 ft/sec
Time of flight TF = 500 days
Flight Angle HCA = 140°

Inclination i = 0°
Communication distance RC_{max} = 4.5 AU
Communication distance $RC_{intercept}$ = 4 AU

(Launch may be phased with Jupiter to miss by less than 0.25 AU.)

Payload

Experimental payload (+ radiometer)	55
Transmitter (5 watts 21 bits/sec 210' DSIF)	10
Antenna (25 db 3' dia)	5
Data encoder, storage etc.	10
Power supply (100 watts isotope)	100
Batteries	10
Launch correction (less propellant)	6
Attitude control (less propellant)	10
Structure and shielding	50
Spacecraft weight at target area	256
Launch correction propellant	4
Attitude control propellant	40
Spacecraft weight at start of trajectory	300
Effective weight of shroud and adapter	60
Total effective weight of payload	360 $^{-0}_{+5}$ %

Launch Vehicle-Payload Combinations

Atlas-Agena + HE*	= 350 lbs.
Floxed Atlas-Agena + HE	= 470 lbs.
Floxed Atlas-Centaur + LE*	= 320 lbs.
Atlas-Centaur + HE	= 620 lbs.

* HE = kick stage I_{SP} = 455 secs, LE = kick stage I_{SP} = 300 secs.

6.2 Minimum Energy Missions to a Given Latitude

At present, no missions have been performed at more than nominal deviations from the ecliptic plane. The major reason for this can be readily seen from Figure 2 in that the required ideal velocity increases rapidly as the heliocentric latitude of the target increases. However if the only restriction placed on the trajectory is that the spacecraft shall reach a given latitude then use can be made of the absolute, minimum ideal velocity trajectory to that latitude. From Figure 2 it can be seen that the intercept point will always be less than 1 AU from the Sun and the flight times will be short, in the region of 80 to 90 days. These flights will remain at the given latitude for very short periods after which the spacecraft returns to and below the ecliptic plane. However it is possible that these missions can survive two or three consecutive half orbits around the Sun and collect information equally from above and below the ecliptic plane.

6.2.1 Minimum Energy Trajectory to 13.5°

Figure 9 shows a typical minimum energy trajectory to $\beta = 13.5°$ which has been used to derive the 13.5° mission profile of Table 6. There will be very little problem with communications since the Earth-spacecraft separation is less than 0.25 AU at intercept. In fact the spacecraft will travel slightly ahead of the Earth with an orbital period of about 320 days - this separation being just sufficient to keep the spacecraft out

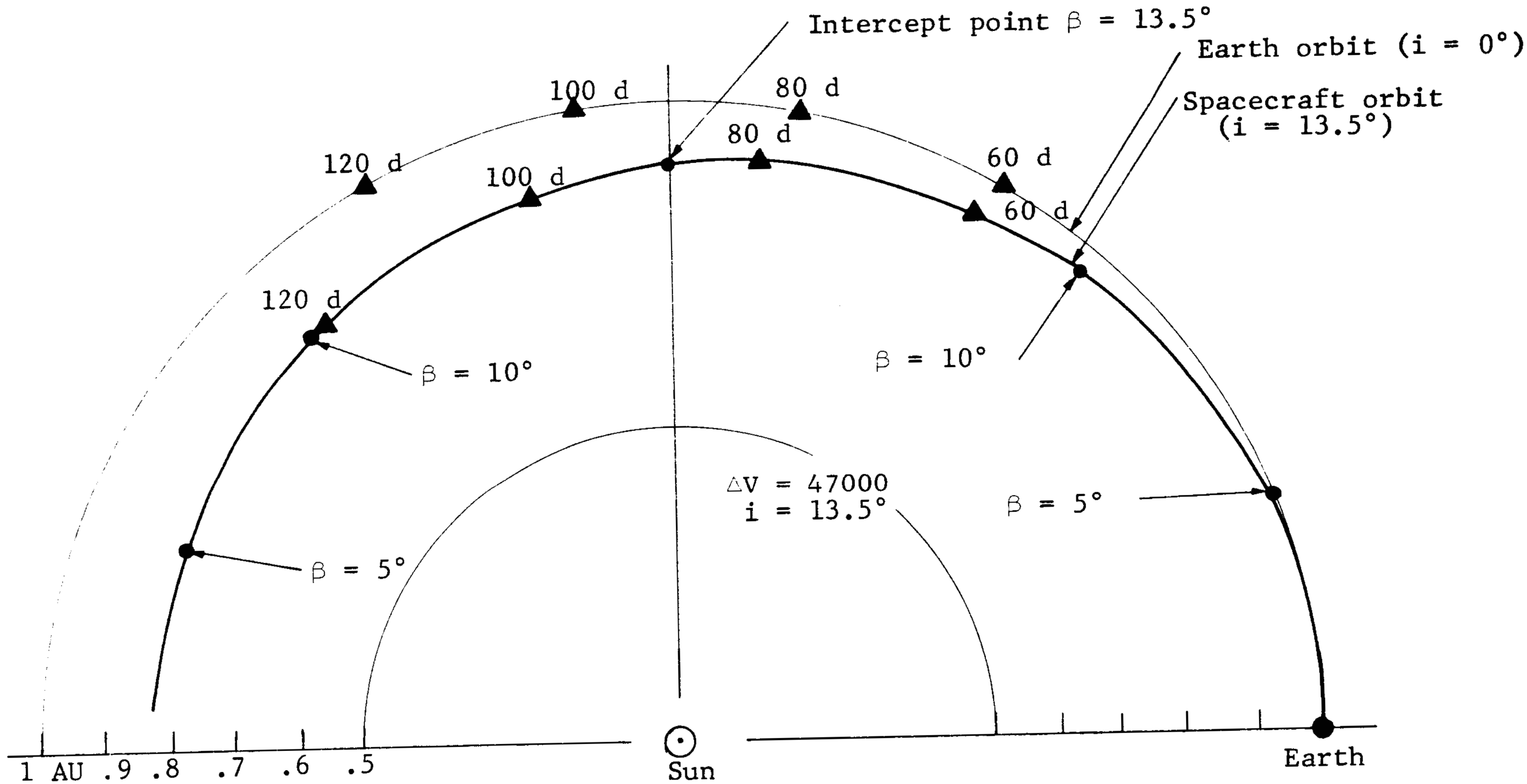


Fig. 9 MINIMUM ENERGY TRAJECTORY TO $\beta = 13.5°$ ($r = 0.9$ AU)

Table 6

INTERPLANETARY MINIMUM ENERGY MISSION TO 13.5° LATITUDE
(r = 0.9 AU)

Trajectory Parameters

Ideal Velocity ΔV = 47,000 ft/sec	Inclination i = 13.5°
Time of flight TF = 88 days	Communications distance
Flight angle HCA = 90°	$RC_{intercept}$ = 0.25 AU

Payload

Experimental payload (+ solar radiometer)	55 lbs.
Transmitter (2 watts 21 bits/sec 85' DSIF)	5
Antenna (17 db 1.5' dia)	3
Data encoder storage etc.	10
Power supply (50 watts solar cell)	40
Batteries	10
Launch correction (less propellant)	7
Attitude control (less propellant)	5
Structure and shielding	40
Spacecraft weight at target area	175
Launch correction propellant	3
Attitude control propellant	10
Spacecraft weight at start of trajectory	188
Effective weight of shroud and adapter	50
Total effective weight of payload	$240^{-0}_{+5}\%$

Launch Vehicle-Payload Combinations

Thor-Agena D + HE*	= 300 lbs.
Atlas-Agena + LE*	= 450 lbs.
Floxed Atlas-Centaur	= 250 lbs.
Titan III-C	= 300 lbs.

* HE = kick stage I_{SP} = 455 secs, LE = kick stage I_{SP} = 300 secs.

of the Earth-Sun line when it recrosses the ecliptic plane. The communications distance will then be 0.4 AU.

6.2.2 Minimum Energy Trajectory to 22°

A second mission profile is included in Table 7 for a minimum energy trajectory to a latitude $\beta = 22°$ out of the ecliptic (Figure 10). Again the flight time will be short ($\simeq$ 90 days) and data should be available from at least one complete orbit of the spacecraft around the Sun (period of orbit $\simeq$ 310 days). The communications distance will be less than 0.5 AU at the first intercept with a latitude of 22° and again the spacecraft will just lead the Earth when it crosses the ecliptic plane.

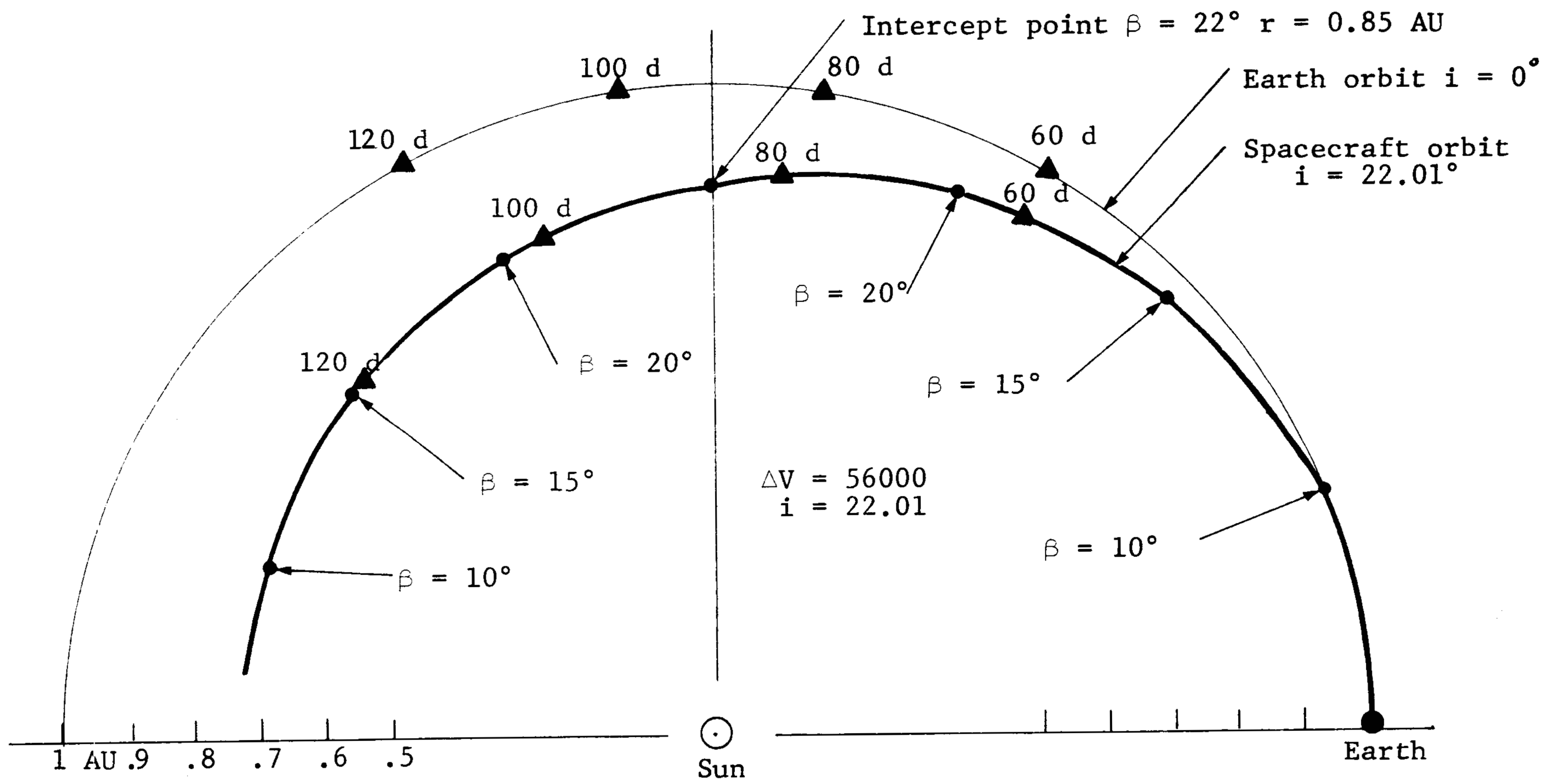


Fig. 10 MINIMUM ENERGY TRAJECTORY TO β = 22° (r = 0.85 AU)

Table 7

INTERPLANETARY MINIMUM ENERGY MISSION TO 22° LATITUDE

(r = 0.85 AU)

Trajectory Parameters

Ideal velocity ΔV = 56,000 ft/sec

Time of flight TF = 87 days

Flight angle HCA = 90°

Inclination i = 22.01°

Communications distance $RC_{intercept}$ = 0.5 AU

Payload

Experimental payload (+ solar radiometer)	55 lbs.
Transmitter (4 watts 21 bits/sec 85'DSIF)	8
Antenna (17 db 1.5' dia)	3
Data encoder, storage etc.	10
Power supply (75 watts solar cell)	50
Batteries	10
Launch correction (less propellant)	7
Attitude control (less propellant)	5
Structure and shielding	45
Weight at target area	193
Launch correction propellant	3
Attitude control propellant	10
Weight at start of trajectory	206
Effective weight of shroud and adapter	50
Total effective weight of payload	260 $^{-0}_{+5}$ %

Launch Vehicle-Payload Combinations

Atlas-Agena + HE*	= 270 lbs.
Atlas-Centaur + LE*	= 200 lbs.
Floxed Atlas-Centaur + LE	= 280 lbs.

* HE = kick stage I_{SP} = 455 secs, LE = kick stage I_{SP} = 300 secs.

6.3 Minimum Energy Missions to a Specified Latitude and Distance

It can be seen from Figure 2 that the required ideal velocity increases quite rapidly if the heliocentric distance at intercept increases or decreases from the optimum value. Thus in specifying a target distance as well as latitude the penalty is paid in ideal velocity. Figure 11 shows the details of a trajectory for a flight to 3 AU and $\beta = 15°$ and Table 8 gives the mission profile. The time of flight will be about 270 days and the spacecraft will be between a latitude of 15° and 17.2° for about 200 days of this time, i.e., between about 1.5 and 3 AU. This is an attractive exploratory mission to non-ecliptic space. The communications distance reaches 4 AU by the time of intercept which will occur about a month before the spacecraft becomes partially obscured by the Sun. This mission will pass above the main concentration of the asteroid belt and amongst other data should provide a good indication of the density gradient of asteroidal dust out of the ecliptic plane. This mission has been specified for the minimum ideal velocity to 3 AU and $\beta = 15°$ and it should be noted that an increase or decrease of the flight time would increase the required ideal velocity. However it will require a Saturn 1B with a Centaur launch vehicle and a low energy ($I_{SP} = 300$) stage. Ballistic flights to larger distances or higher latitudes would require a Saturn V type of vehicle.

However it does not seem reasonable to use a Saturn V with preferably three or more stages to place a relatively small spacecraft ($\simeq$ 500 lbs.) beyond 3 AU and β = 15°. A far better alternative would be to use a thrusted stage which could be launched with a Saturn 1B vehicle. Preliminary calculations of thrusted trajectories indicate that with a $J \simeq 25$ (J is the thrusted equivalent of I_{SP}) and flight times limited to 300 days, all heliocentric latitudes are accessible. The distance-latitude combinations range from about 4 AU in the ecliptic through 2 AU at 30° to 0.5 AU at 90°. To put the parameter J in perspective it can be noted that the SNAP 50 system should develop a $J \simeq 50$ and be feasible beyond about 1975. By increasing the time of flight constraint for thrusted trajectories larger heliocentric distances to all latitudes become accessible.

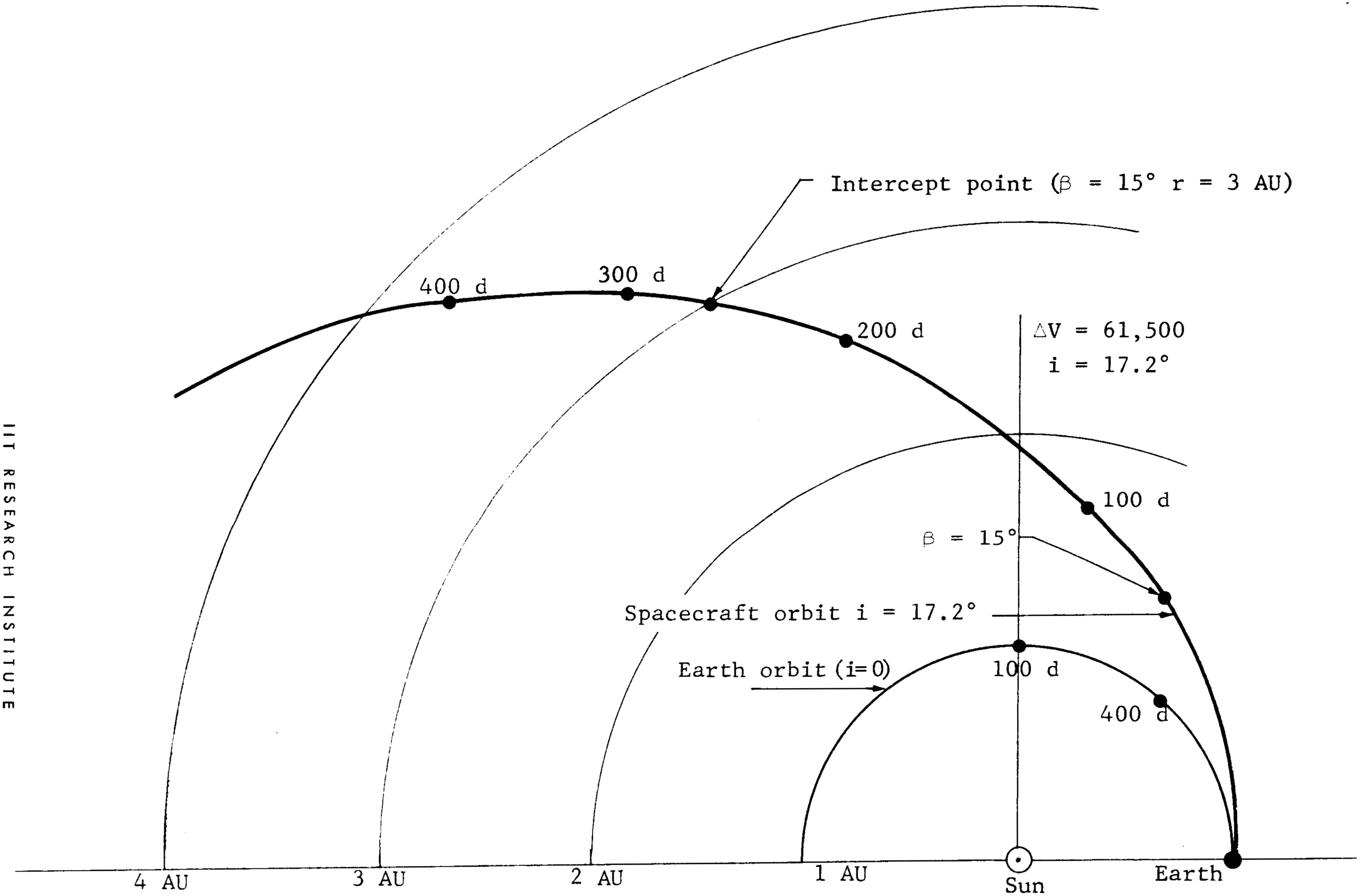


Fig. 11 MINIMUM ENERGY TRAJECTORY TO β = 15° r = 3 AU

Table 8

INTERPLANETARY MINIMUM ENERGY MISSION TO 3 AU, 15° LATITUDE

Trajectory Parameters

Ideal velocity ΔV = 61,500 ft/sec
Time of flight TF = 265 days
Flight angle HCA = 118°

Inclination i = 17.2°
Communications distance $RC_{intercept}$ = 4 AU
Time above β = 15° = 200 days

Payload

Experimental payload (high capacity dust detector)	55
Transmitter (5 watts 25 bits/sec 210' DSIF)	10
Antenna (25 db 3' dia)	6
Data encoder, storage etc.	10
Power supply (100 watts isotopic)	100
Batteries	10
Launch correction (less propellant)	6
Attitude control (less propellant)	10
Structure and shielding	50
Spacecraft weight at target area	257
Launch correction propellant	4
Attitude control propellant	25
Spacecraft weight at start of trajectory	286
Effective weight of shroud and adapter	60
Total effective weight of payload	350 $^{-0}_{+5}$ %

Launch Vehicle-Payload Combinations

Floxed Atlas-Centaur + HE*	= 250 lbs.
Saturn 1B + Centaur + LE*	≃ 400 lbs.
Titan III-C (minus trans stage) + Delta II + LE	≃ 900 lbs.

* HE kick stage I_{SP} = 455 secs, LE kick stage I_{SP} = 300 secs.

6.4 Multiple Missions

It is difficult to present a generalized method for determining the optimum trajectories which will place two or more spacecraft in predetermined relative positions in space. However a few examples are given here of possible multiple mission configurations.

The most obvious combination of trajectories is a highly elliptical Earth orbiting trajectory in conjunction with an interplanetary trajectory. The Earth orbital spacecraft monitors the interplanetary medium for about 1/4 of its orbit near apogee. The other spacecraft monitors the medium along its trajectory. This provides measurements at two points in space for comparison although for separations of the order of an AU between the spacecraft there will be a significant longitude separation.

Another configuration is to support a planetary spacecraft with a second interplanetary mission such that a desirable spatial alignment is achieved during a critical phase of a planetary mission - probably during planet intercept. The trajectories in this case cannot be generalized since the planetary trajectory will be specific for each launch date. However for a particular launch window it would be possible to calculate the optimum supporting trajectories for the second spacecraft.

Where there is no specific target for either of the missions it is possible to use the generalized trajectories discussed in Section 4 to determine the flight parameters for multiple interplanetary missions in the ecliptic plane. For ecliptic flights it is possible to use a simple rotation to align longitudes but it becomes rather complex to align longitude and latitude since each flight would require a significantly different inclination (Appendix 1). Figures 12-15 have been included to show the launch separation and ideal velocities for alignment of three spacecraft at a given time after the first launch. Figure 12 shows the trajectories for alignment at 5 AU, 4 AU and 3 AU after 500 days. Similar curves (Figures 13 and 14) are shown for 300 and 200 day alignment in the ecliptic plane. Finally Figure 15 shows the relative positions of spacecraft using the same ideal velocity and trajectory but just separating the launch date.

Mission profiles have not been tabulated for multiple missions since their major objectives will probably only be definable after the single exploratory missions have been interpreted.

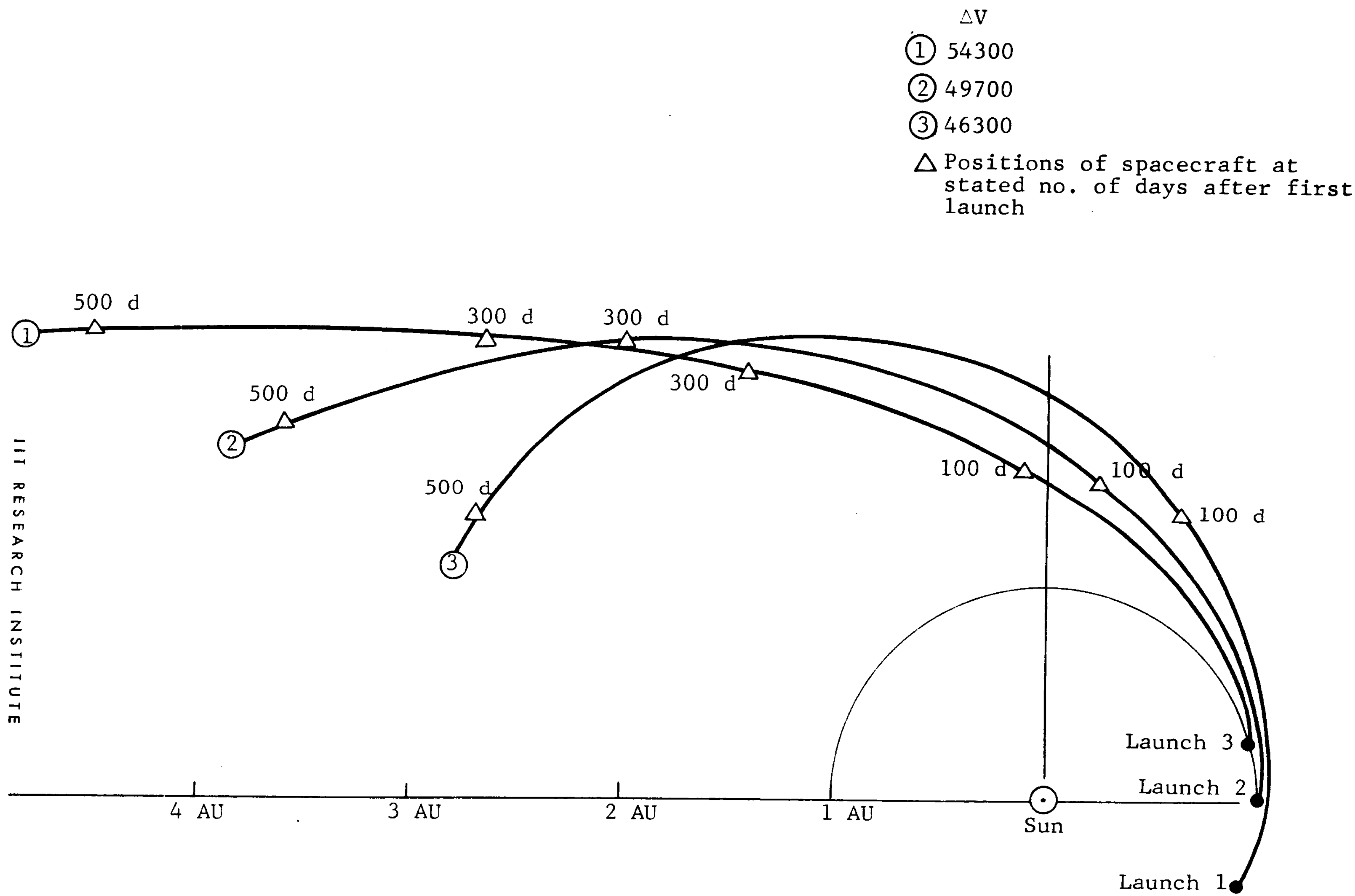


Fig. 12 EXAMPLE OF MULTIPLE MINIMUM ENERGY TRAJECTORIES IN ECLIPTIC PLANE (ALIGNMENT WITH SUN AFTER 500 DAYS AT 3 AU, 4 AU AND 5 AU)

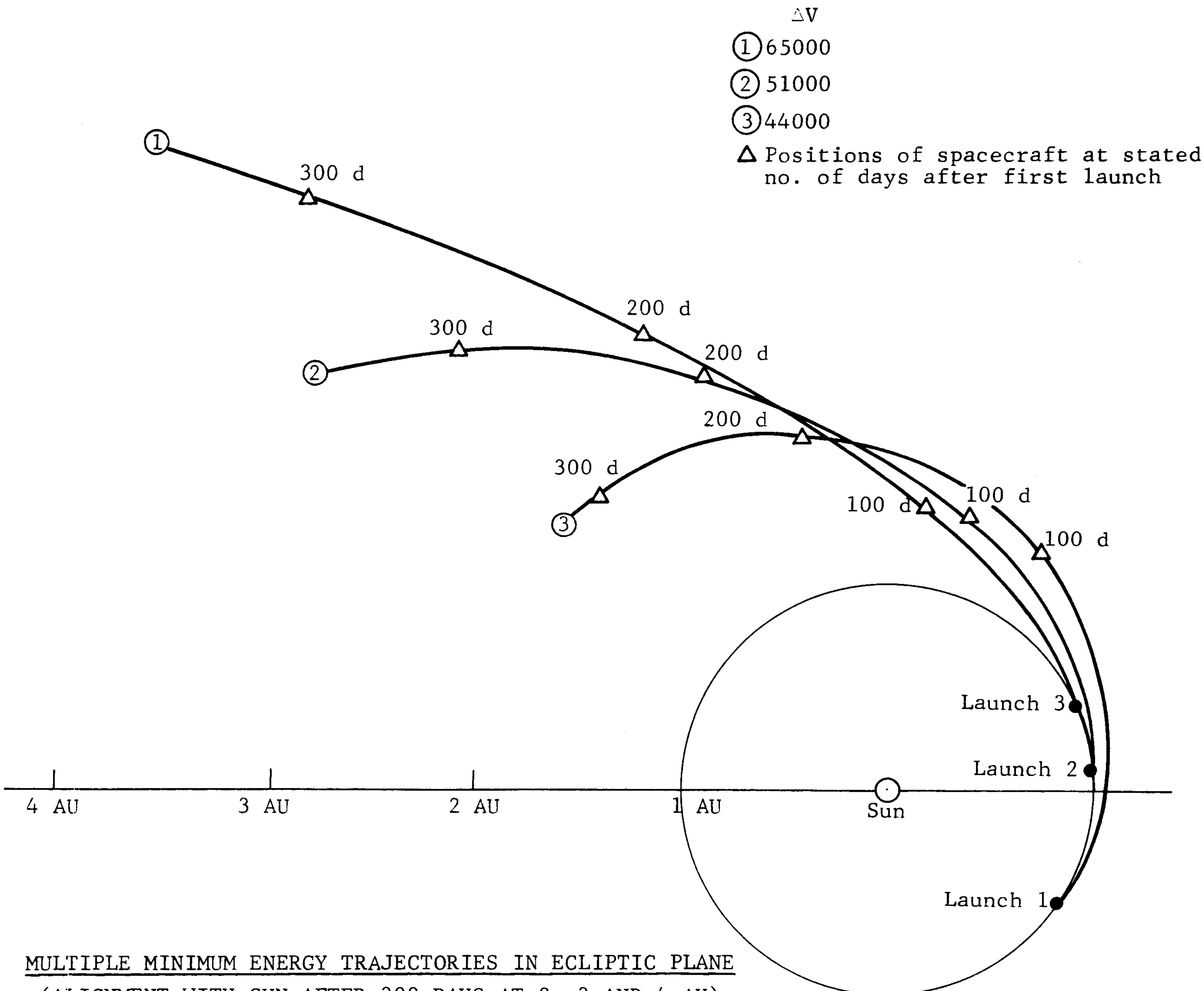


Fig. 13 MULTIPLE MINIMUM ENERGY TRAJECTORIES IN ECLIPTIC PLANE (ALIGNMENT WITH SUN AFTER 300 DAYS AT 2, 3 AND 4 AU)

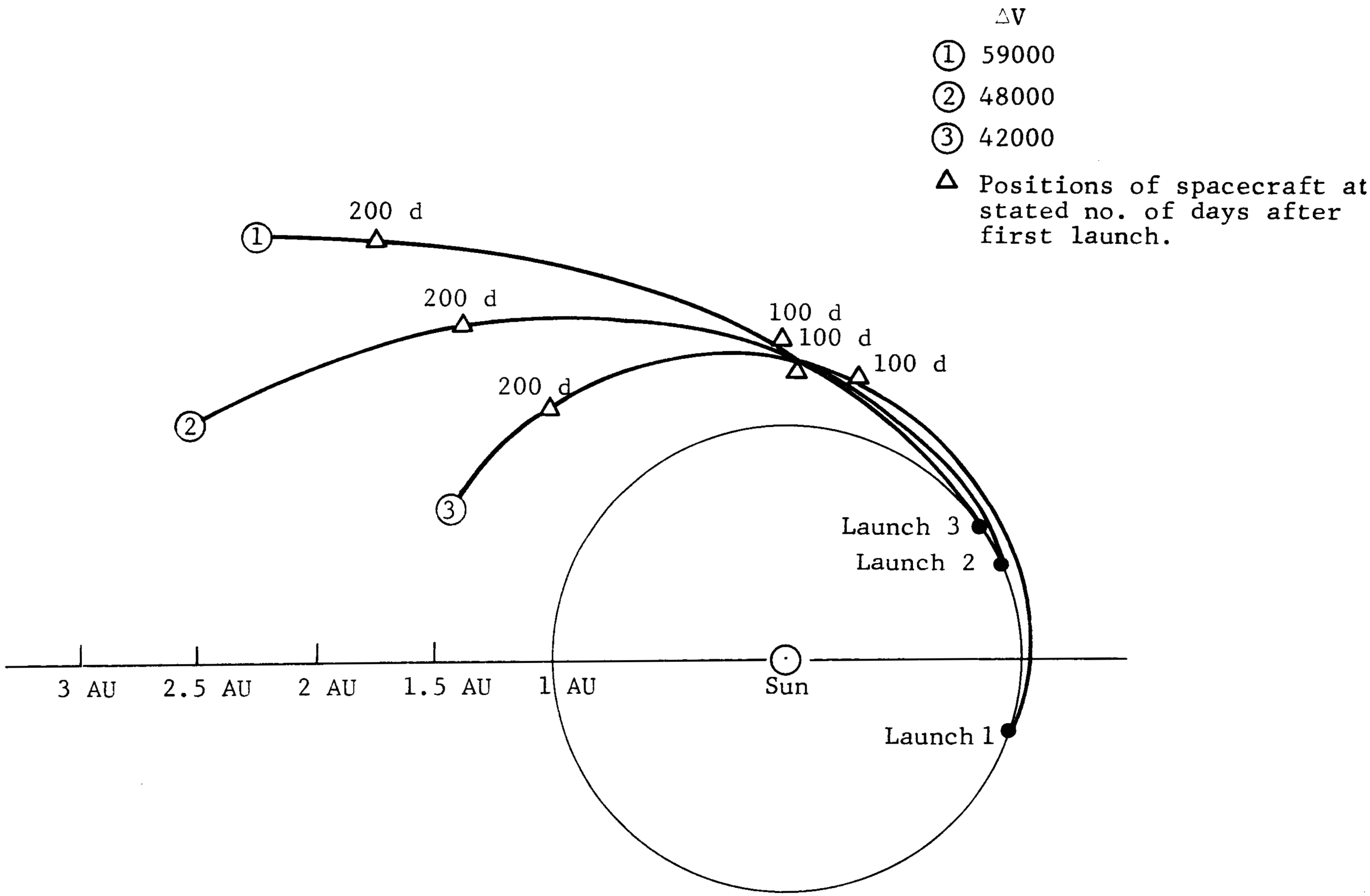
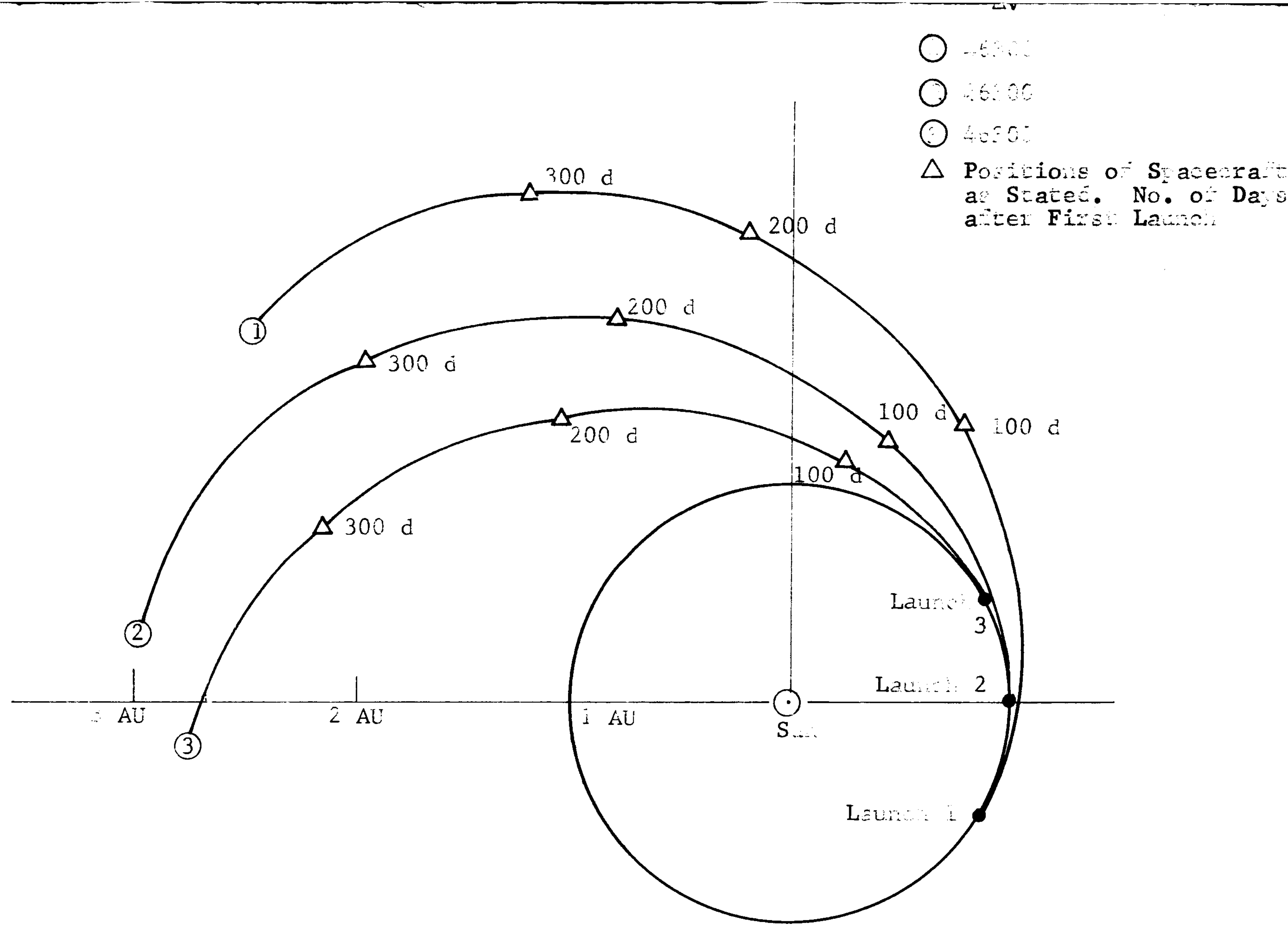


Fig. 14 MULTIPLE MINIMUM ENERGY TRAJECTORIES IN ECLIPTIC PLANE (ALIGNMENT WITH SUN AFTER 200 DAYS AT 1.5, 2, 2.5 AU)



Fi . 15 MULTIPLE MINIMUM ENERGY TRAJECTORIES IN ECLIPTIC PLANE
(HOHMANN TRANSFERS TO [illegible] AU WITH LAUNCH DELAY OF [illegible]0 AND [illegible]0 DAYS)

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APPENDIX 1

FAMILIES OF MINIMUM ENERGY TRAJECTORIES

The families of trajectories cover the region of space within 5 AU of the Sun and between $\pm$ 50° latitude. All trajectories are drawn as they would appear in the orbit plane.

APPENDIX 1

FAMILIES OF MINIMUM ENERGY TRAJECTORIES

1. INTRODUCTION

It is of great convenience when studying future deep space missions to have available a generalized compilation of trajectory data. This appendix presents such a set of data which includes not only the most obvious parameters such as ideal velocity and flight time but also heliocentric flight angle, communications distance, relative Earth-Sun and spacecraft positions, inclination of the orbit plane and time above the target latitude, all for minimum energy trajectories to a specified heliocentric distance (r) and latitude (β). This is possible when no heliocentric longitude is used in the specification of the target area. Therefore the compilation has limited application for planetary missions where the longitude of the Earth and the target are constrained by their respective ephemerides. However for interplanetary missions where a relative Earth-target longitude is not specified they offer a good first approximation to the required parameters.

2. MINIMUM ENERGY TRAJECTORIES IN THE ECLIPTIC PLANE

The most familiar minimum energy trajectory in the ecliptic plane is the Hohmann transfer. This requires a launch at the perihelion of the orbit for trajectories outside the Earth's orbit (aphelion inside 1 AU) with a flight angle of 180°.* For a Hohmann trajectory the velocity at launch is parallel to the Earth's velocity (V_e) and hence maximum assistance is obtained from V_e. Figure A1 shows a family of perihelion launched trajectories which are Hohmann ellipses where aphelion is less than infinity (Curves 1-5). (Also shown is the perihelion-launched parabolic trajectory (Curve 6) and hyperbolic trajectories (Curves 7-12).) Time of flight contours are included.

An interesting situation arises, however, in that although it is clear that these are minimum energy trajectories to points at their aphelia, they are also very close to the minimum energy trajectories to all points in the stated time of flight. This approximation arises because the heliocentric velocity required by a spacecraft is a maximum at the perihelion of an orbit and it is sometimes better to launch slightly after perihelion where a lower spacecraft velocity is demanded and which more than compensates for the slight loss of assistance from the Earth's orbital velocity. The separation between perihelion and launch rarely exceeds a few degrees for minimum energy trajectories in the ecliptic plane. Thus Figure A-1 gives a family of trajectories from which one can be selected

* Only trajectories outside 1 AU will be discussed since those inside evolve directly from the same reasoning and substitution of aphelion for perihelion.

which is a close approximation to the minimum ideal velocity trajectory to a specified distance, r, in a specified time of flight. For example, to arrive at 3 AU in 310 days will require an ideal velocity 48,500 ft/sec, the heliocentric flight angle will be 140°, and the communications distance at intercept will be 4 AU with the spacecraft obscured by the Sun. A similar set of trajectories is shown in Figure A2 for flights inside the Earth's orbit.

3. MINIMUM ENERGY TRAJECTORIES OUT OF THE ECLIPTIC PLANE

The same type of presentation has been used for non-ecliptic trajectories, where the target area is specified by a distance and a latitude. The trajectories are drawn as they appear in the orbit plane. Families of trajectories have been drawn for values of β, the target latitude, of 5°, 10°, 15°, 20°, 25°, 30°, 40°, and 50°. However it is possible to interpolate for trajectories between these discrete values.

A feature of non-ecliptic trajectories is that if the orbit plane is inclined at i°, the heliocentric latitude of consecutive points along the orbit varies from 0° to i° to 0° to -i° etc. In an orbit plane of inclination i, lines of constant latitude β can be drawn through the Sun at angles $\pm\ \alpha$ and 180° $\pm\ \alpha$ to the Earth-Sun line at launch. The angle α is given by

$$\sin \alpha = \frac{\sin \beta}{\sin i}$$

When $\beta > i$ the orbit does not include any points at latitude β; when $\beta = i$ then $\alpha = 90°$ and just one line of constant β exists.

When $0 < \beta < i$ then four such lines exist in the orbit plane, two above and two below the ecliptic plane. Thus in general there are just four points on an elliptical orbit of inclination i which have a latitude $\beta < i$. The two of these below the ecliptic plane need not be considered because of the symmetry of the orbit. Of the other two, one will be at a significantly larger heliocentric distance, usually in the second quadrant of the orbit. By specifying β and i a single locus of intercept points is defined and by further specifying a distance, r, a single intercept point is defined with respect to the Sun and the Earth at launch. The minimum energy trajectory to this point can be found for that orbit plane. If an orbit plane of different inclination i', say, is considered, the target point (r, β) does not have the same longitude but is still specifically defined, and another minimum ideal velocity trajectory can be found for the i' plane. There is clearly an inclination i_o which provides a truly a minimum energy trajectory to each specified target (r, β). In fact these trajectories will also not be launched at perihelion for the same reason as for the ecliptic trajectories in that the lower velocity required at points after perihelion more than compensates for the loss of assistance from the Earth's orbital velocity. This is accentuated for non-ecliptic missions since a component of V_e is already lost by launching at an angle to the ecliptic plane. In fact for non-ecliptic trajectories the angle between perihelion and launch is approximately equal to the inclination angle.

Figures A3 to A18 show families of minimum energy trajectories to heliocentric distances between 0.5 and 5 AU for the specified target latitudes. The trajectories are drawn as they would appear in the orbit plane. It can be noted that very few of the orbit planes have inclinations equal to the desired latitude. Since in general $i > \beta$, the spacecraft will travel above the target latitude for part of its orbit. This is shown by the full line part of the trajectory and for low inclination large distance flights this is for the major part of each half orbit. The locus of intercept points for each latitude is shown and therefore ideal velocity, time of flight and orbit inclination can be estimated for intermediate trajectories. Communication distances can also be estimated remembering that the orbit is inclined to the Earth's orbital plane. By observing all the minimum energy trajectories to a given latitude β, the absolute minimum ideal velocity trajectory to that β can be deduced and in fact will always intercept at a point inside the Earth's orbit. This can also be seen from the plots of minimum ideal velocity in Figure 2 of the preceding report.

4. MULTIPLE TRAJECTORIES

An interesting additional use of the families of trajectories lies in the construction of multiple missions. The ecliptic plane family has been used as an example. Since the curves will always be minimum energy whatever the relative position of the Earth at launch, the figure can be rotated through 360° to cover all points in the ecliptic plane. Thus a series of target points can be selected at various radii and, say, in alignment with the Sun after 500 days from the first launch (Fig. 12). The minimum energy trajectories can be determined as the ones that pass through the respective distances with the correct time of flight. Rotation of the basic family of trajectories about the Sun is required to achieve this but then yields the appropriate launch separation. Since there is only one minimum energy trajectory to a given radius in a specified time of flight, each of the multiple trajectories will have different ideal velocities and flight parameters.

This technique is restricted for non-ecliptic trajectories and only very approximate results can be obtained. This is because each of the trajectories is only minimum energy to the target point and unlike the ecliptic trajectories do not represent the minimum energy flights to other points along them. Clearly the latitudes of the other points are not equal to β. In view of the fact, however, that the inclinations are only one or two degrees larger than β they can be used for very approximate estimations of multiple trajectories to various radii aligned with the Sun.

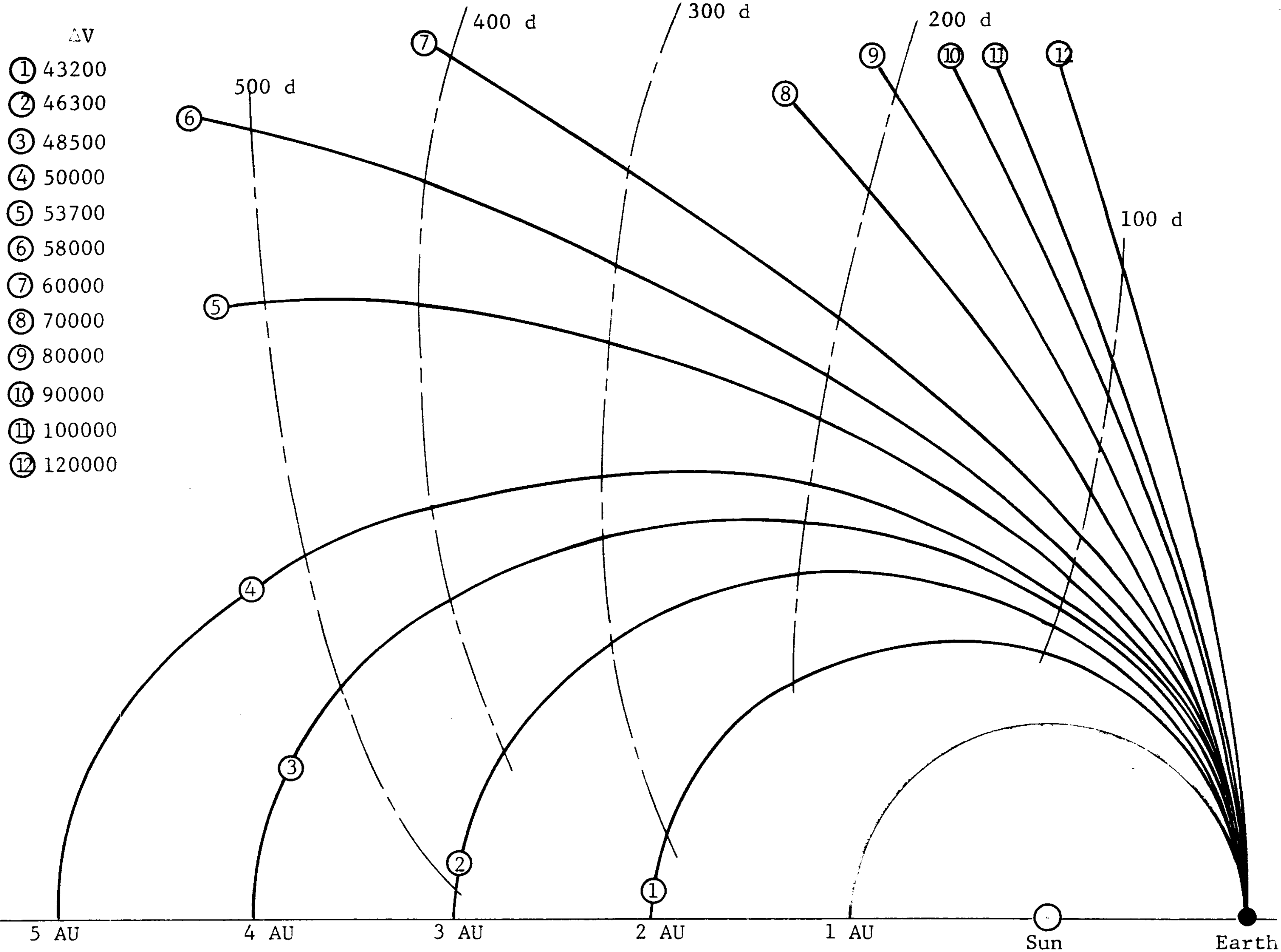

Fig. A1 FAMILY OF MINIMUM ENERGY ECLIPTIC TRAJECTORIES

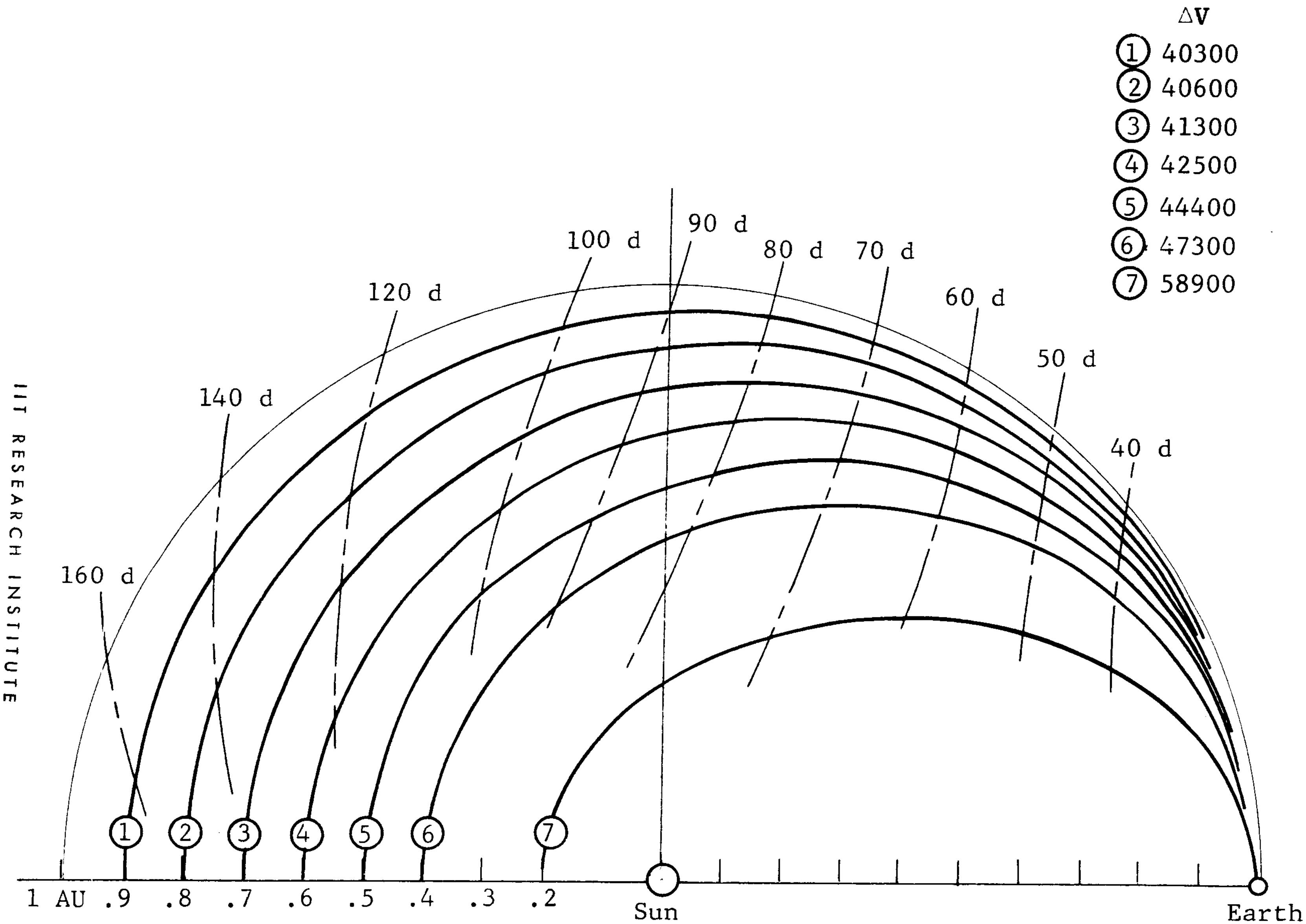

Fig. A2 FAMILY OF MINIMUM ENERGY ECLIPTIC TRAJECTORIES

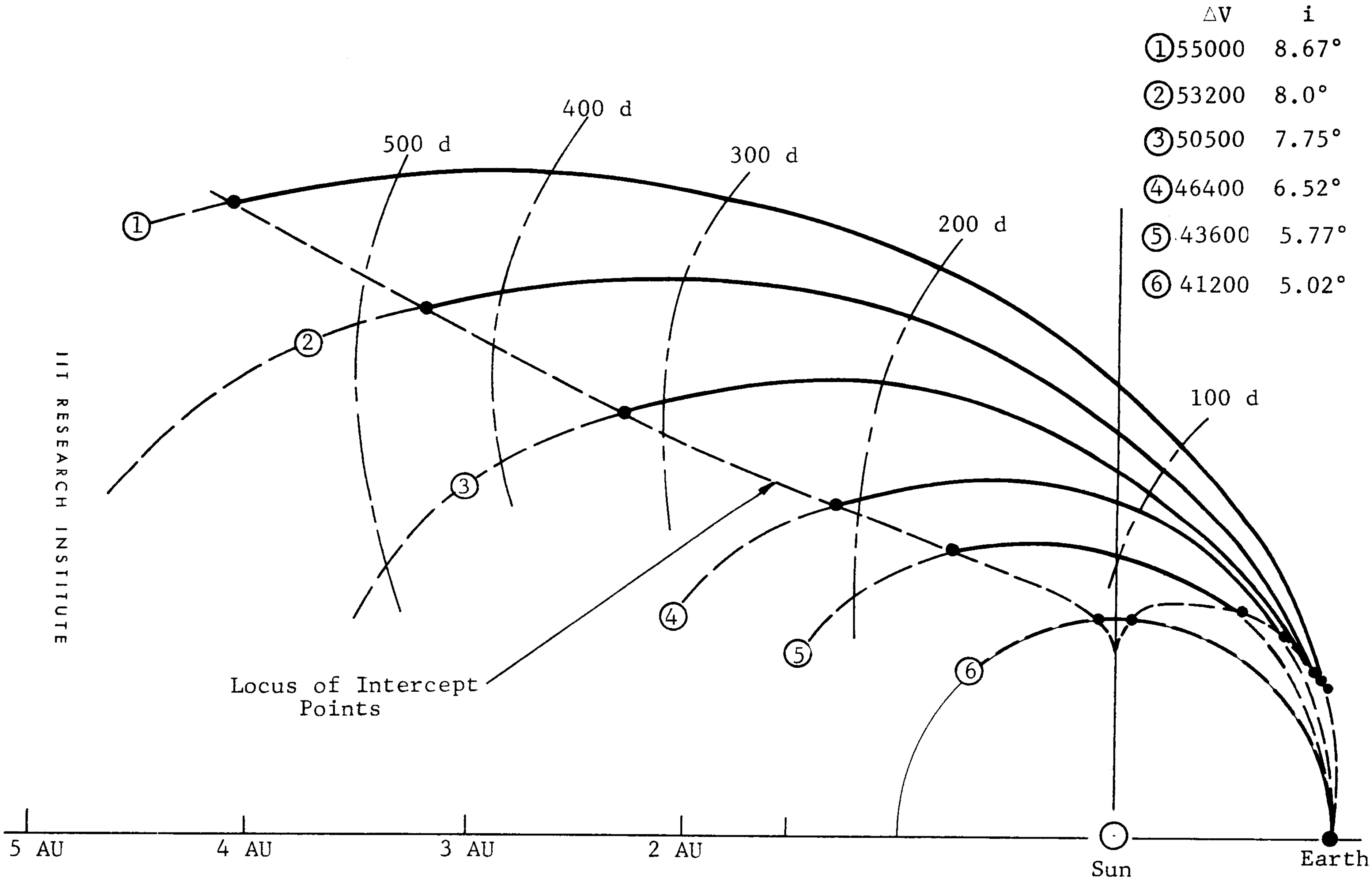

Fig. A3 FAMILY OF MINIMUM ENERGY TRAJECTORIES TO A LATITUDE OF 5° ($1 < r < 5$ AU)
(TRAJECTORIES DRAWN IN PLANE OF ORBIT)

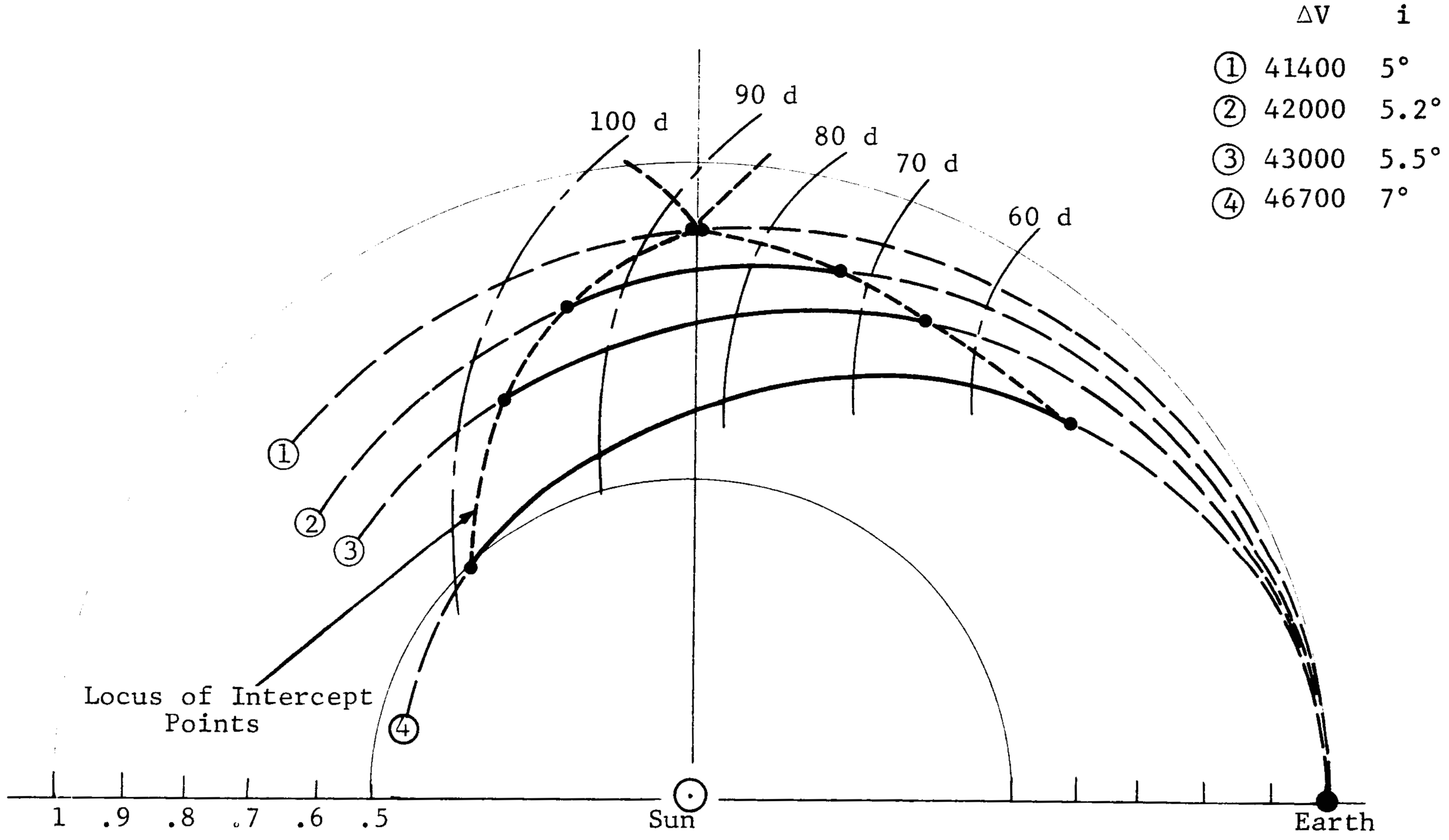

Fig. A4 FAMILY OF MINIMUM ENERGY TRAJECTORIES TO A LATITUDE OF $\beta = 5°$ ($0.5 < r < 1$ AU)
(TRAJECTORIES DRAWN IN PLANE OF ORBIT)

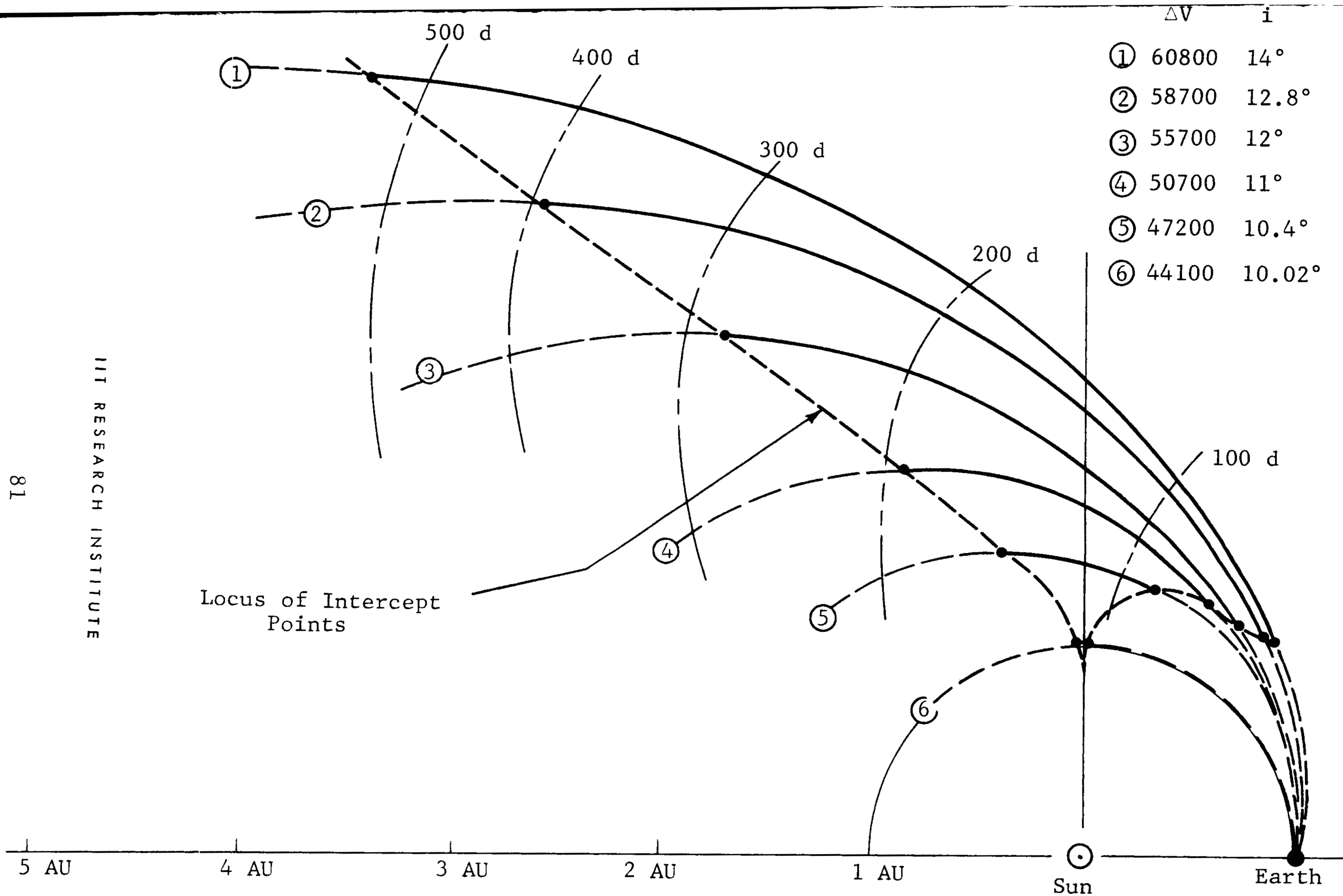

Fig. A5 FAMILY OF MINIMUM IDEAL VELOCITY TRAJECTORIES TO A LATITUDE $\beta = 10°$ $(1 < r < 5$ AU) (TRAJECTORIES DRAWN IN PLANE OF ORBIT)

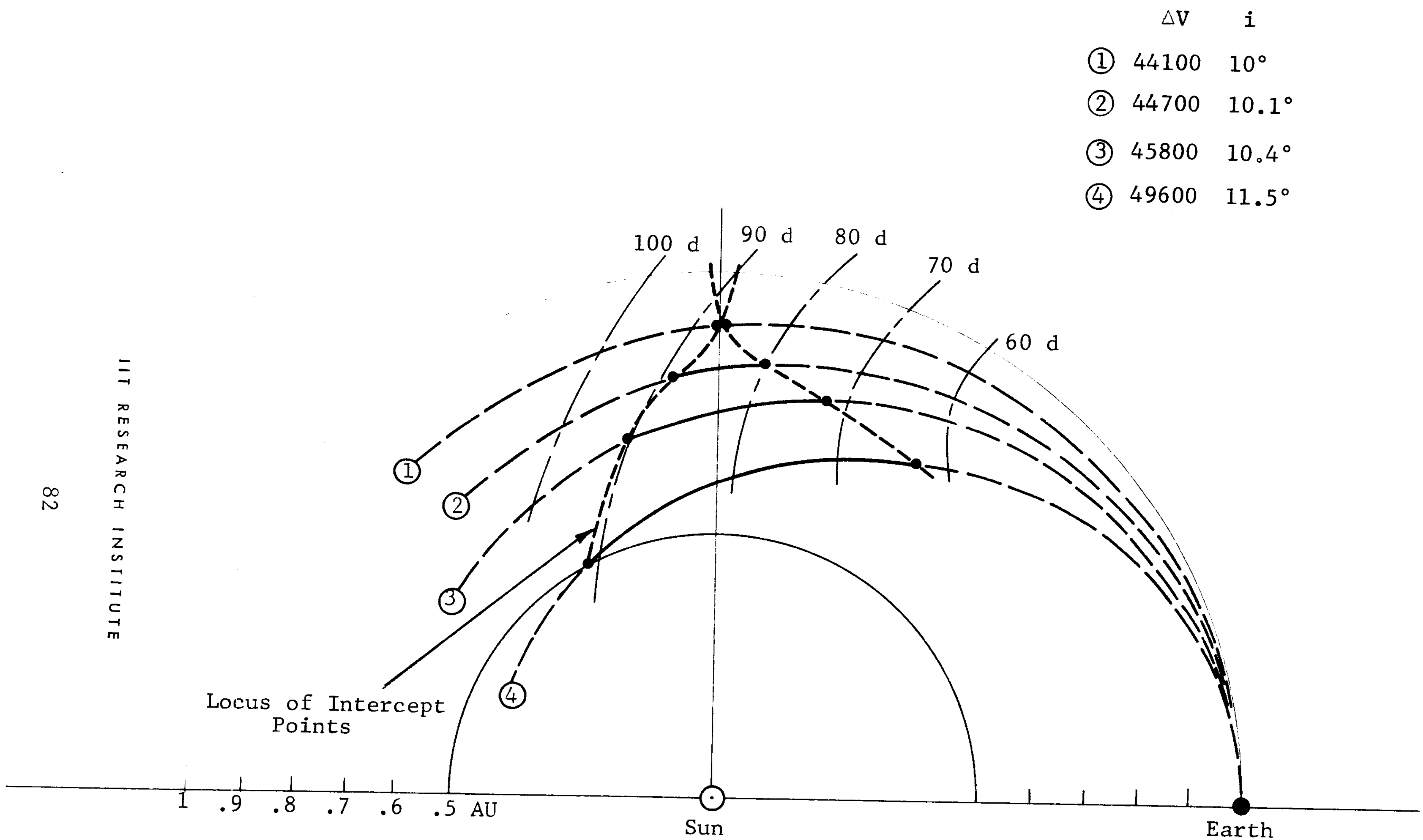

Fig. A6 FAMILY OF MINIMUM IDEAL VELOCITY TRAJECTORIES TO A LATITUDE $\beta = 10°$ $(0.5 < r < 1$ AU) (TRAJECTORIES DRAWN IN PLANE OF ORBIT)

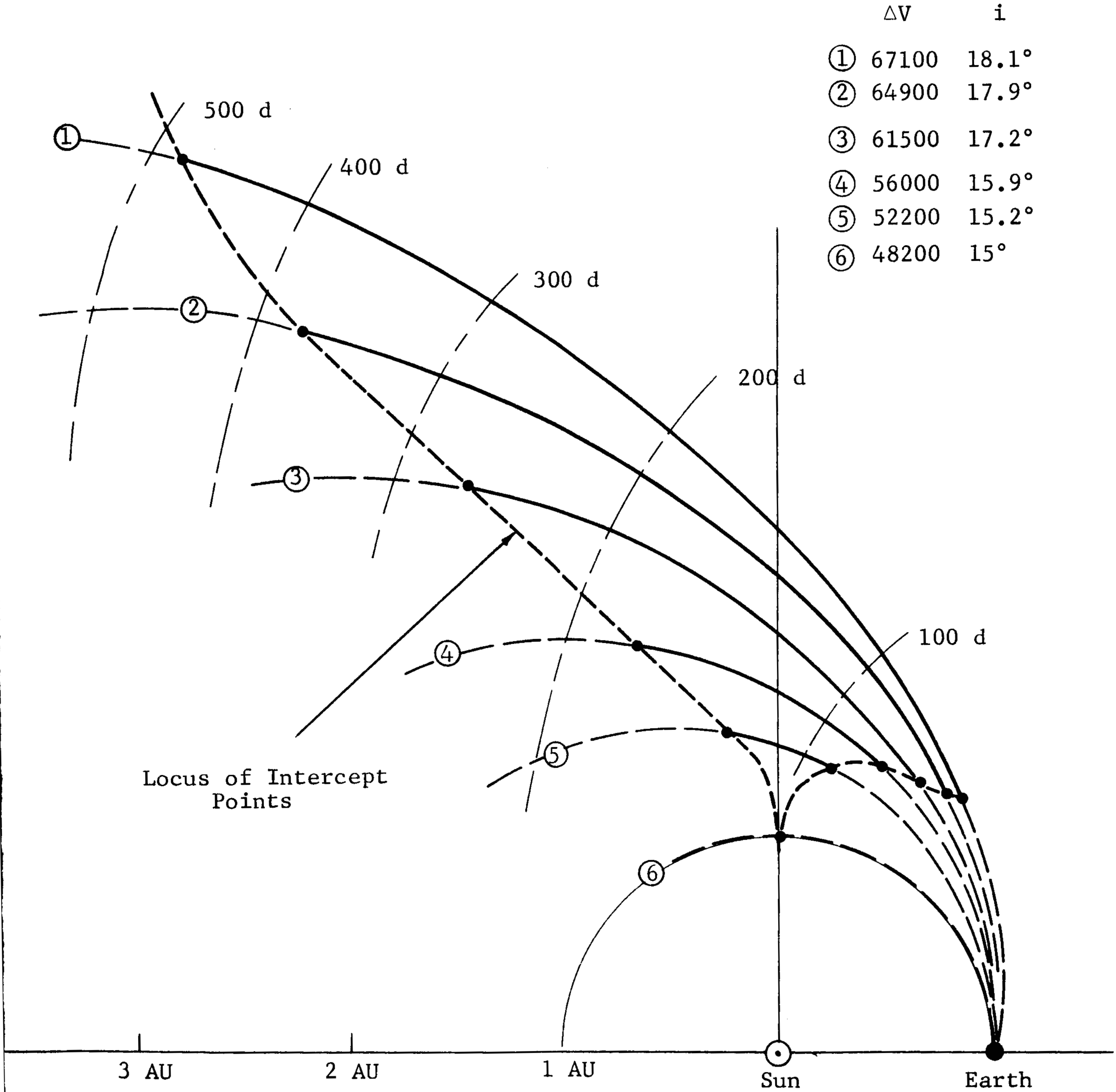

Fig. A7 FAMILY OF MINIMUM IDEAL VELOCITY TRAJECTORIES TO A LATITUDE $\beta = 15°$ ($1 < r < 5$ AU) (TRAJECTORIES DRAWN IN PLANE OF ORBIT)

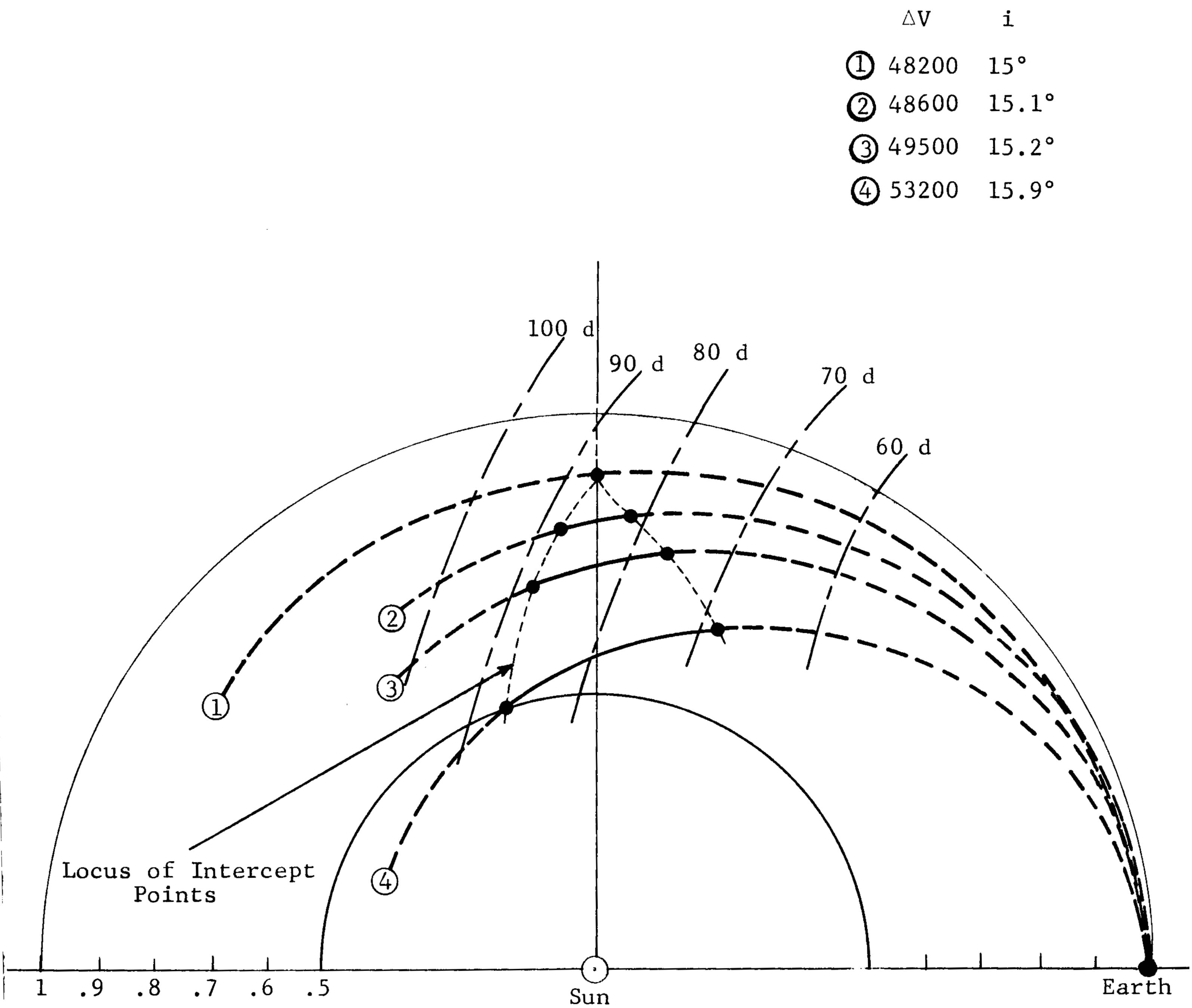

Fig. A8 FAMILY OF MINIMUM ENERGY TRAJECTORIES TO A LATITUDE $\beta = 15°$ ($0.5 < r < 1$ AU)(TRAJECTORIES DRAWN IN PLANE OF ORBIT)

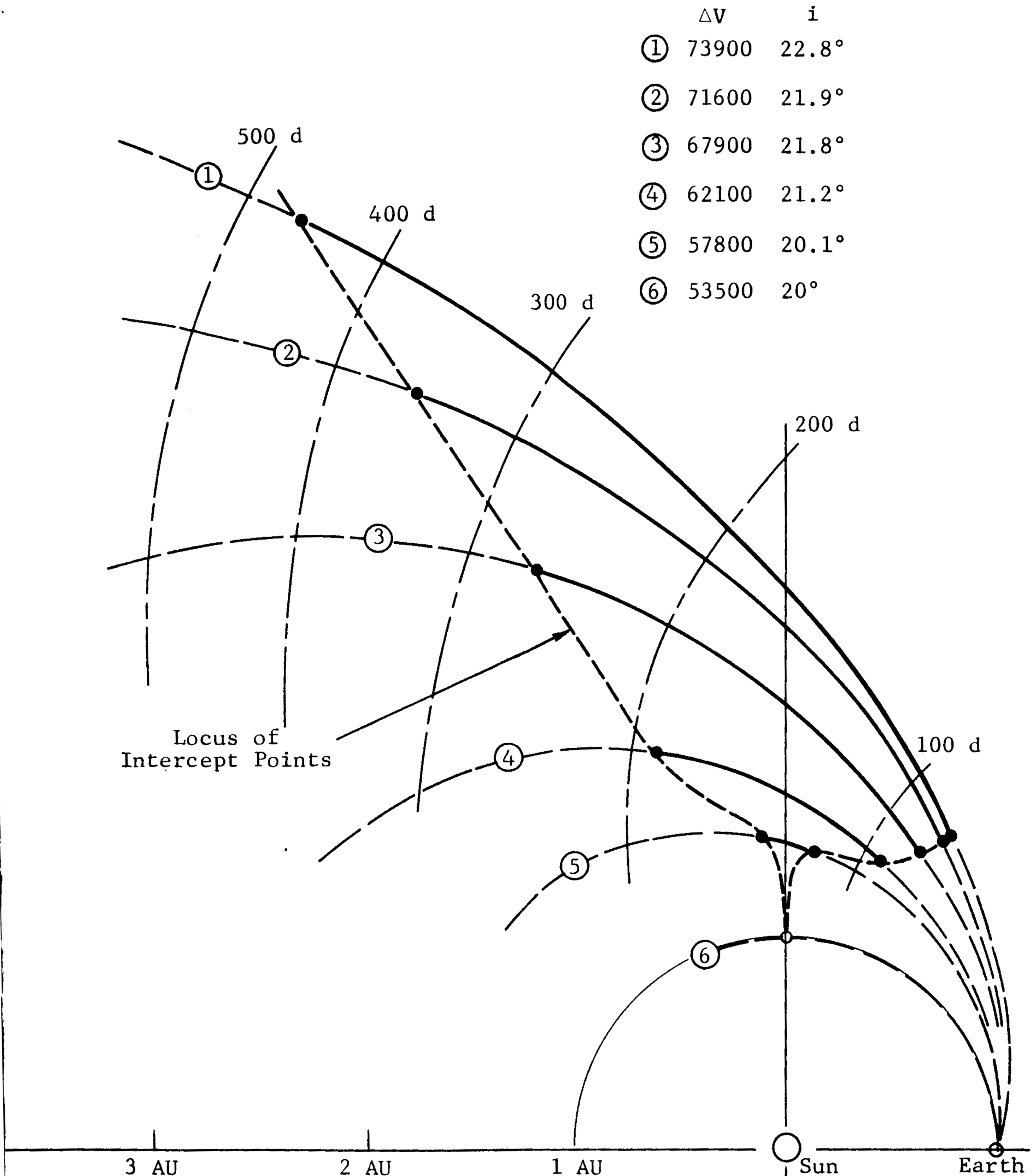

Fig. A9 FAMILY OF MINIMUM ENERGY TRAJECTORIES TO A LATITUDE OF 20° (1 < r < 5 AU) (TRAJECTORIES DRAWN IN PLANE OF ORBIT AT INCLINATION i°)

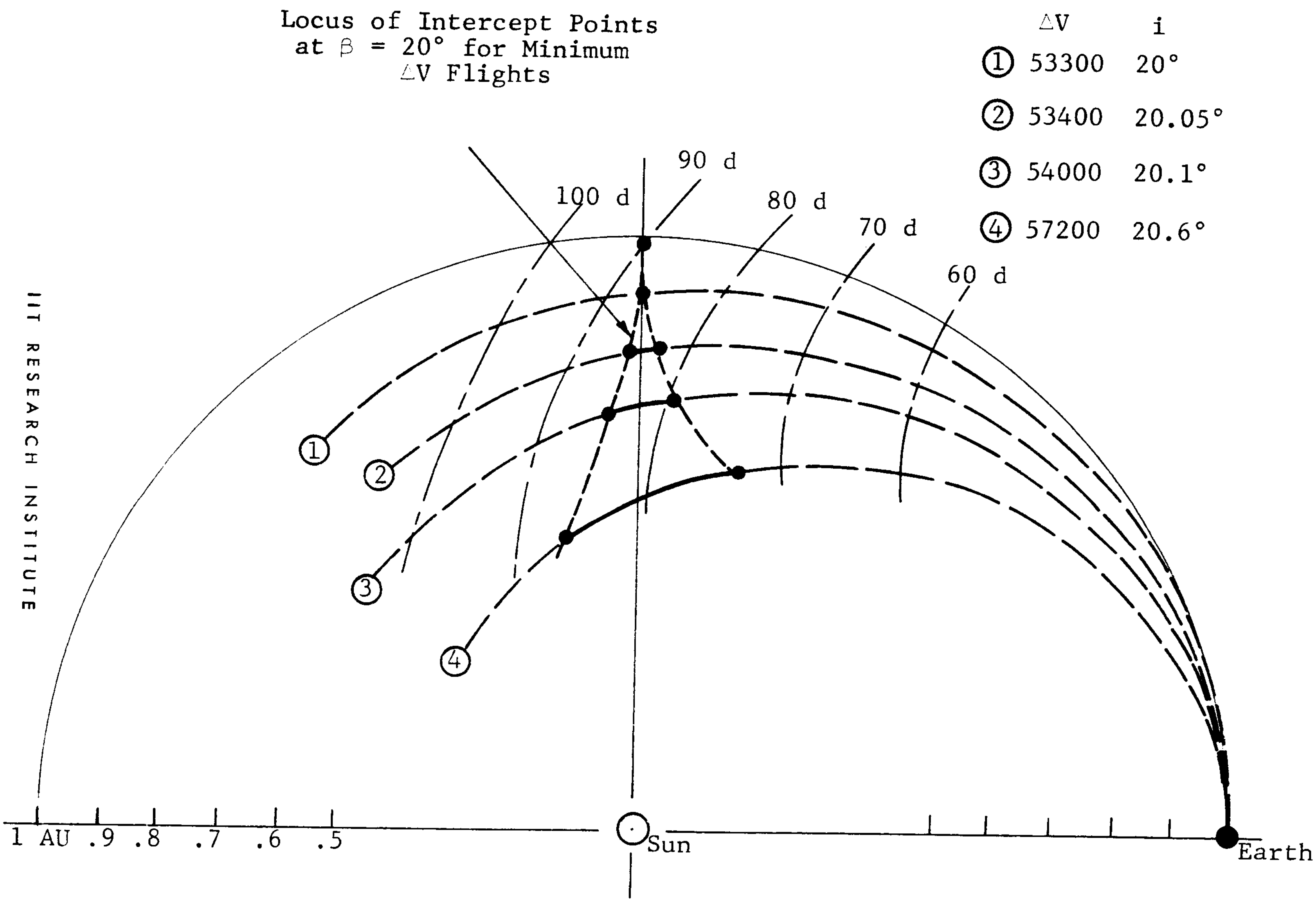

Fig. A10 FAMILY OF MINIMUM ENERGY TRAJECTORIES TO A LATITUDE OF 20° (0.5 < r < 1 AU)
(TRAJECTORIES DRAWN IN PLANES OF ORBIT AT INCLINATION i°)

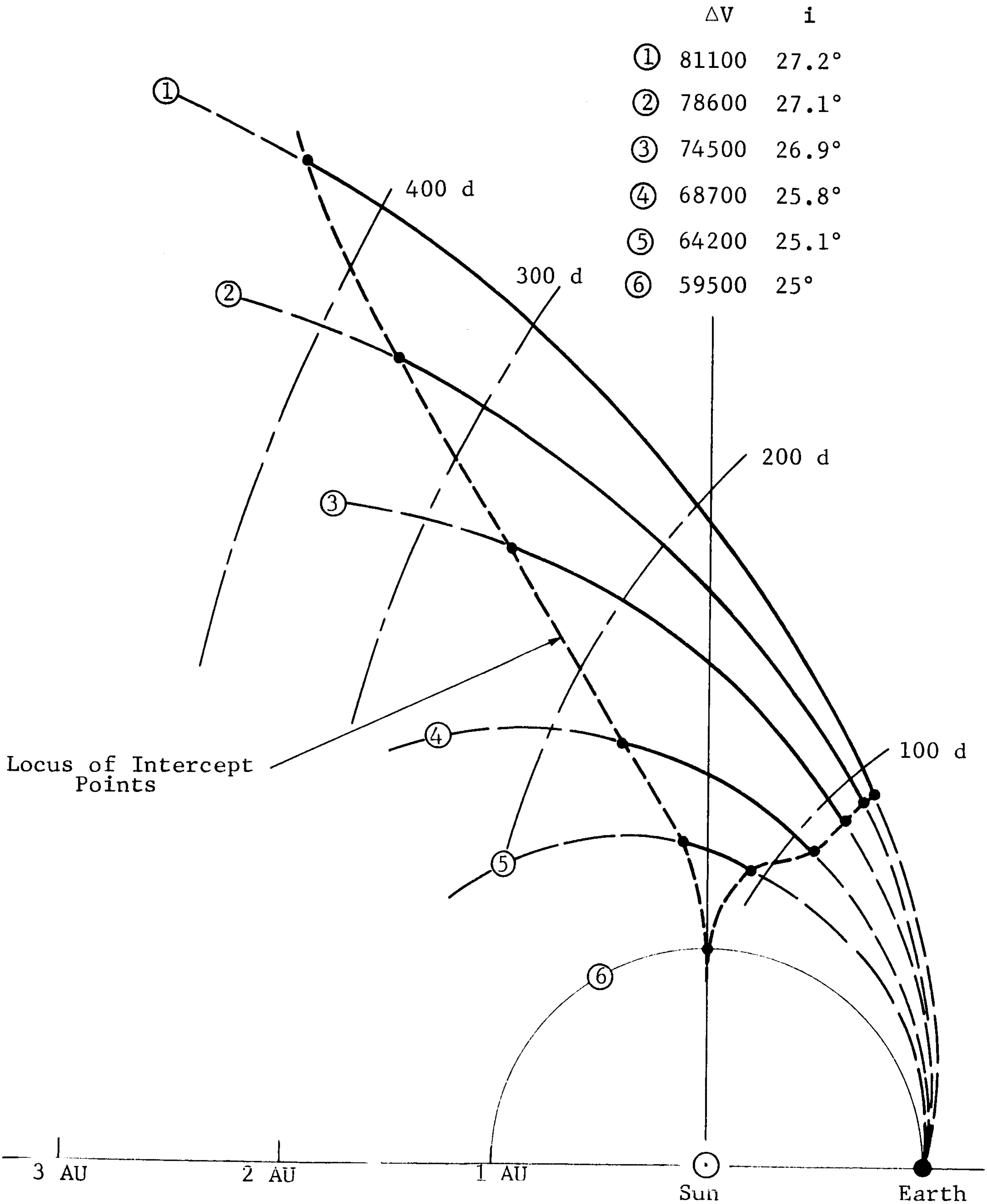

Fig. A11 FAMILY OF MINIMUM IDEAL VELOCITY TRAJECTORIES TO A LATITUDE β = 25° (1 < r < 5 AU) (TRAJECTORIES DRAWN IN PLANE OF ORBIT)

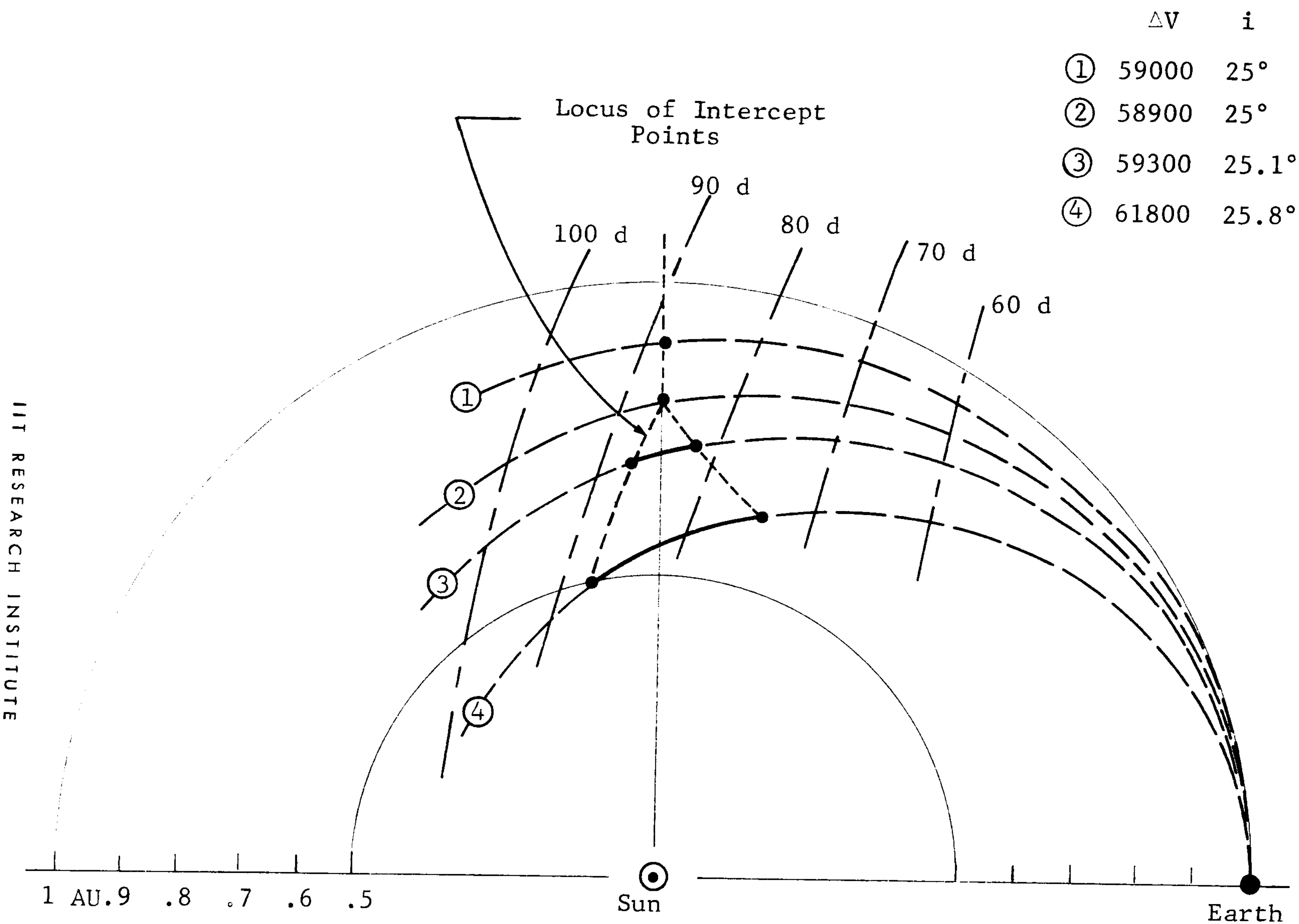

Fig. A12 FAMILY OF MINIMUM ENERGY TRAJECTORIES TO A LATITUDE $\beta = 25°$ ($0.5 < r < 1$ AU)(TRAJECTORIES DRAWN IN PLANE OF ORBIT)

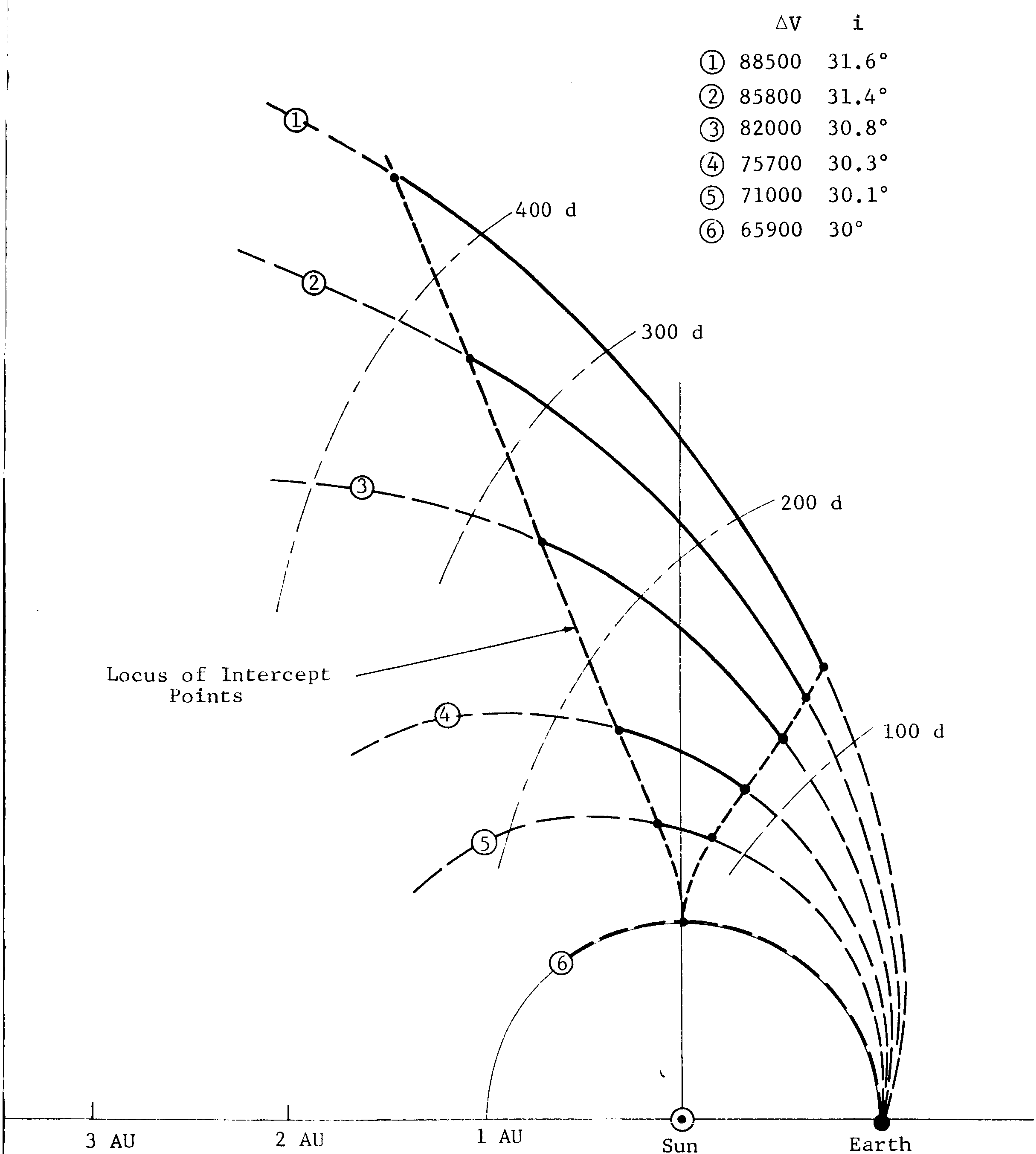

Fig. A13 FAMILY OF MINIMUM IDEAL VELOCITY TRAJECTORIES TO A LATITUDE β = 30° (1 < r < 5 AU)(TRAJECTORIES DRAWN IN PLANE OF ORBIT)

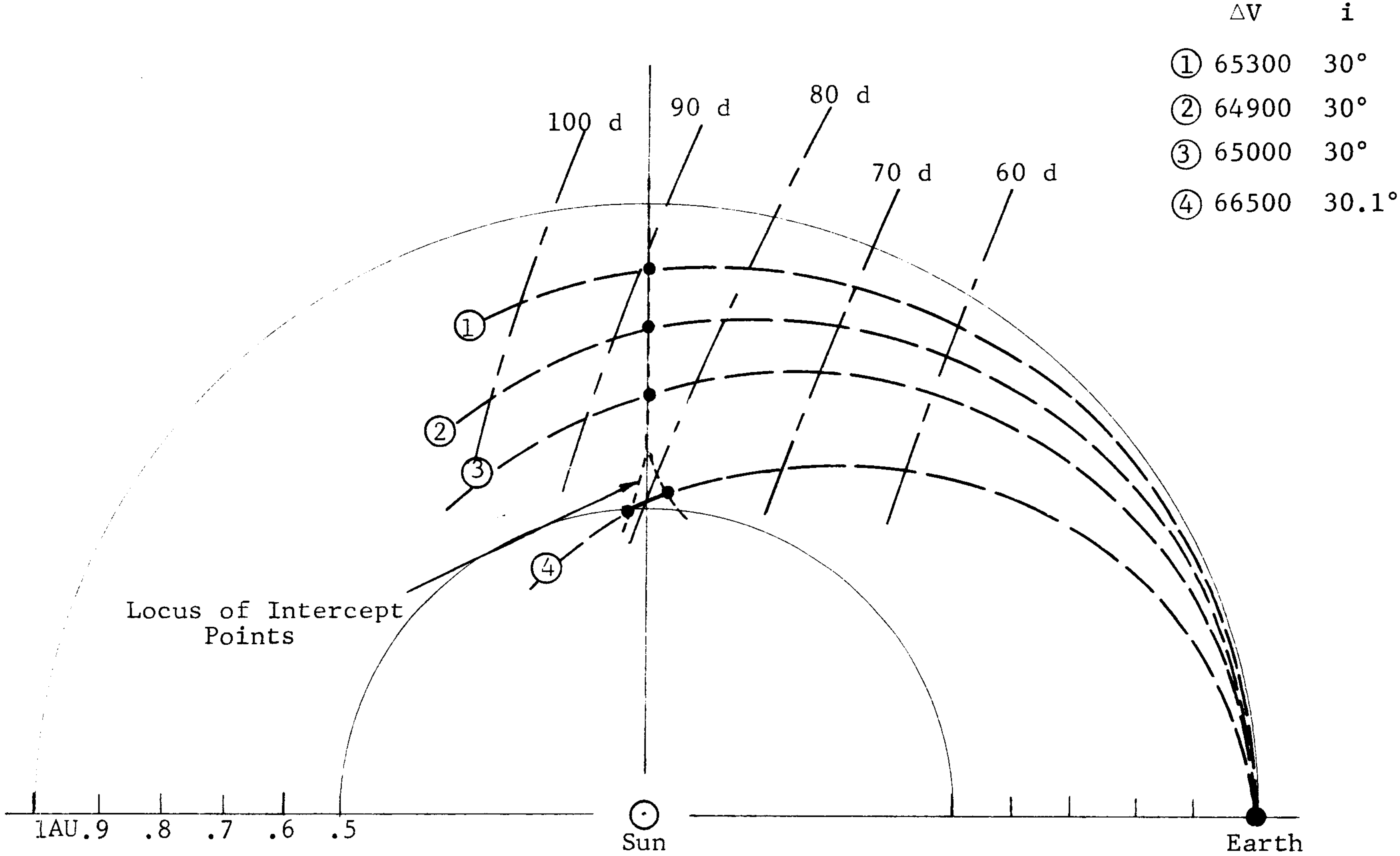

Fig. A14 FAMILY OF MINIMUM IDEAL VELOCITY TRAJECTORIES TO A LATITUDE β = 30° (0.5 < r < 1 AU)(TRAJECTORIES DRAWN IN PLANE OF ORBIT)

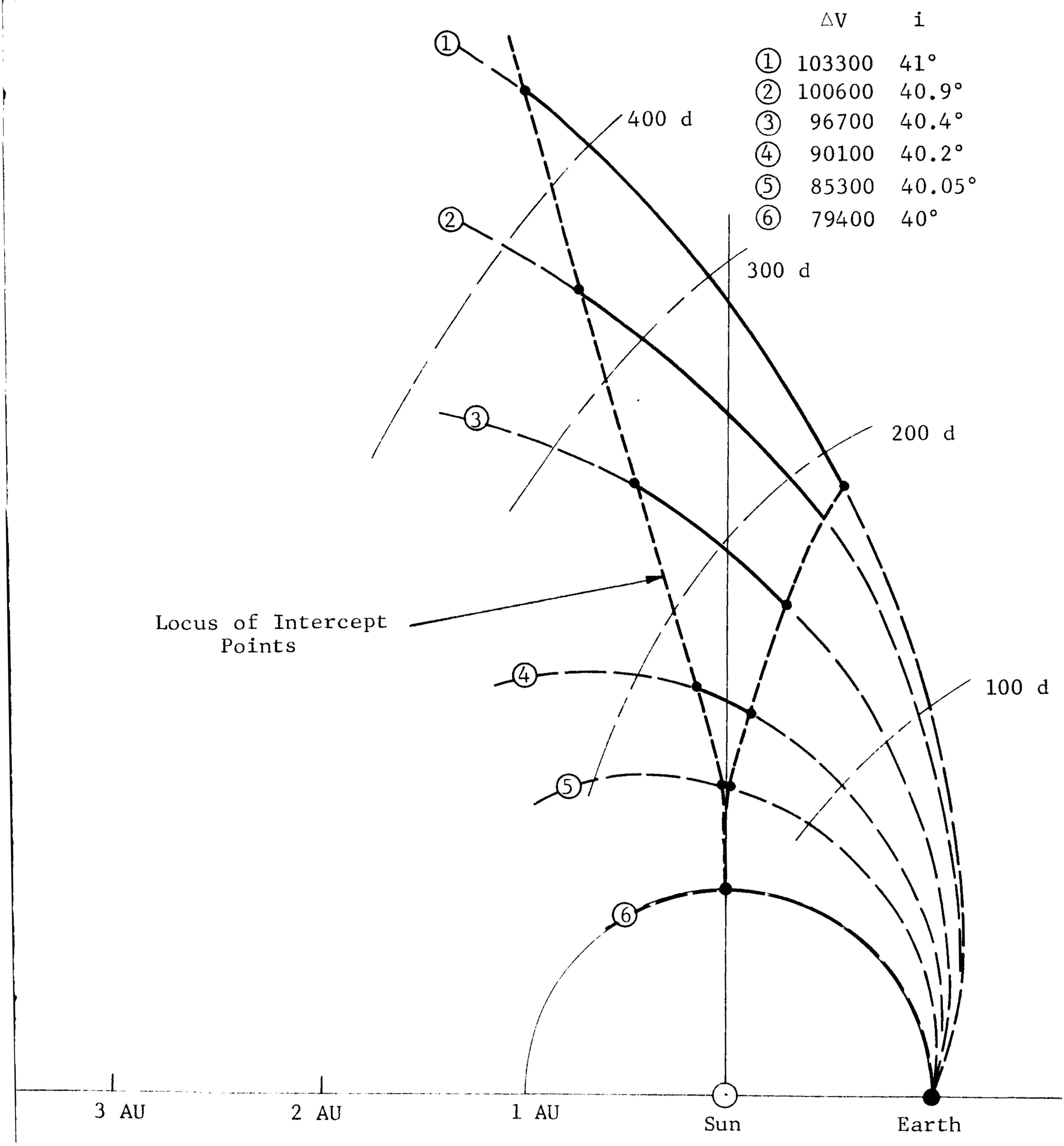

Fig. A15 FAMILY OF MINIMUM IDEAL VELOCITY FLIGHTS TO A LATITUDE $\beta = 40°$ ($1 < r < 5$ AU)(TRAJECTORIES DRAWN IN PLANE OF ORBIT)

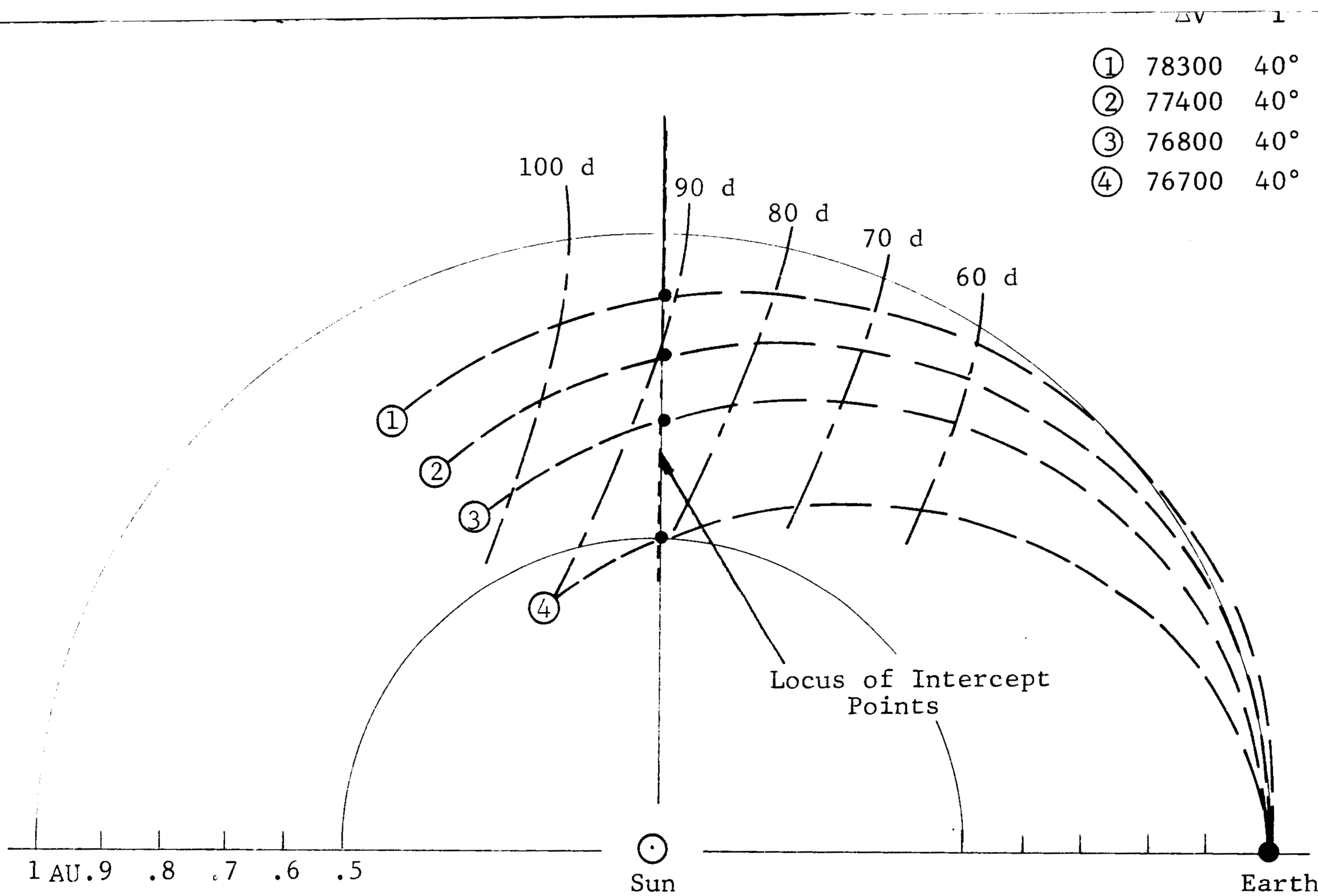

Fig. A16 FAMILY OF MINIMUM IDEAL VELOCITY TRAJECTORIES TO A LATITUDE $\beta = 40°$ $(0.5 < r < 1$ AU) (TRAJECTORIES DRAWN IN PLANE OF ORBIT)

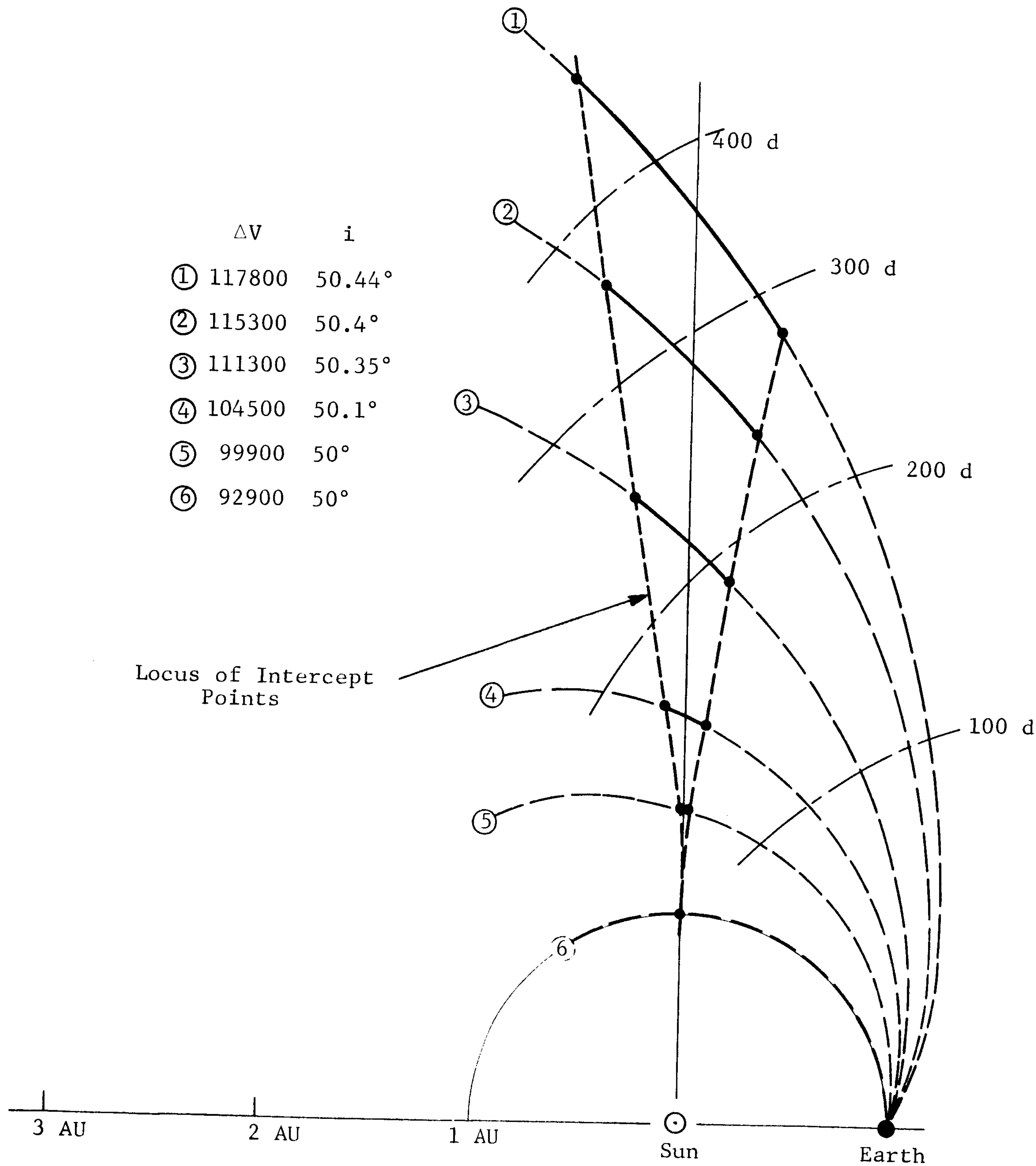

Fig. A17 FAMILY OF MINIMUM IDEAL VELOCITY TRAJECTORIES TO A LATITUDE β = 50° (1 < r < 5 AU)(TRAJECTORIES DRAWN IN PLANE OF ORBIT)

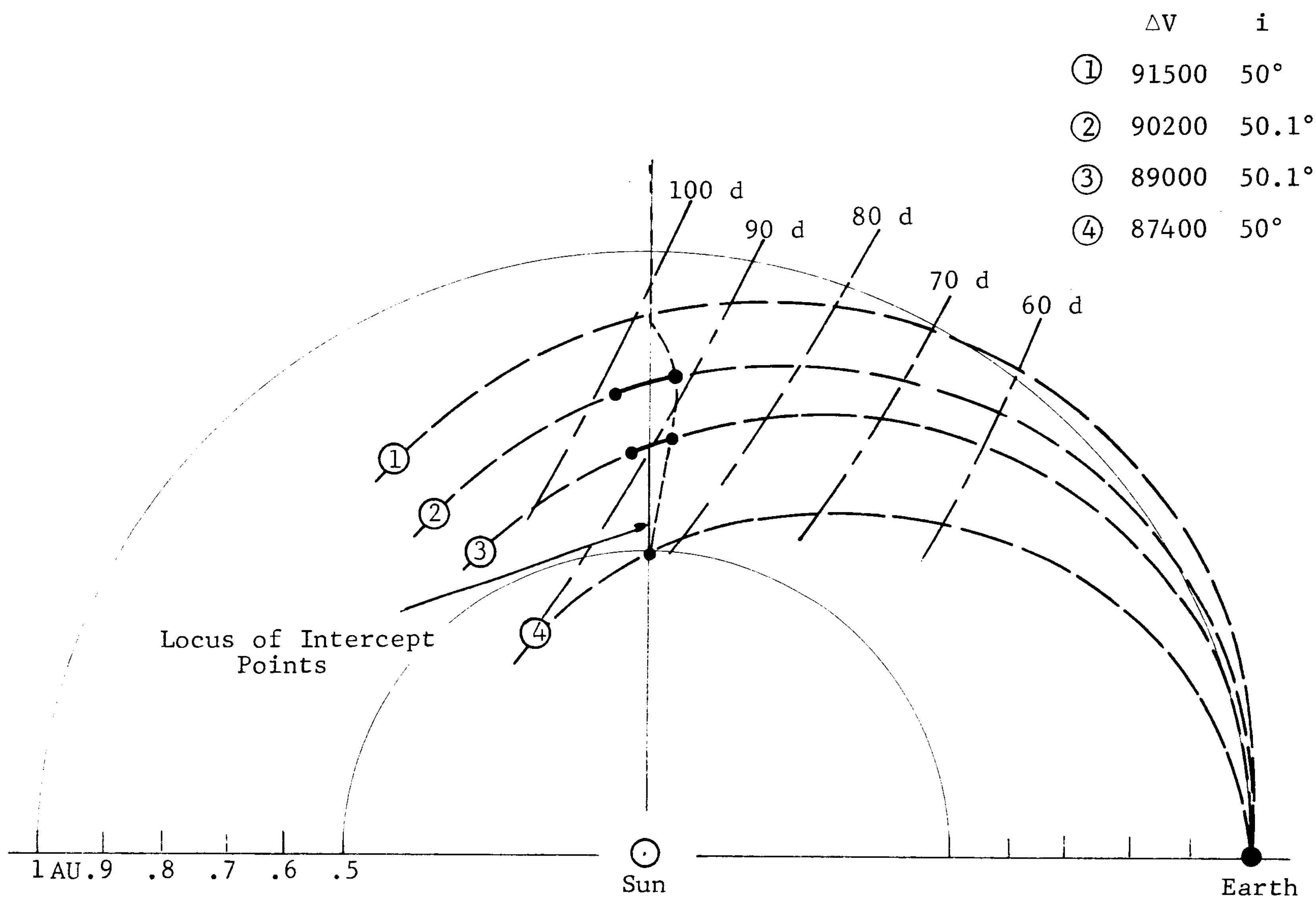

Fig. A18 FAMILY OF MINIMUM IDEAL VELOCITY TRAJECTORIES TO A LATITUDE $\beta = 50°$ $(0.5 < r < 1$ AU) (TRAJECTORIES DRAWN IN PLANE OF ORBIT)